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CONTRACT REQUIREMENTS	CONTRACT ITEM	MODEL	CONTRACT NO.	DATE
Exhibit E	Par. 5.1	LEM	NAS 9-1100	14 Jan.1963

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REPORT

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DATE: 15 May 1963

APPLICATION OF LEM TECHNOLOGY TO NASA LUNAR LANDING RESEARCH PROGRAM [U]

CODE 26512

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GRUMMAN AIRCRAFT ENGINEERING CORPORATION

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TABLE OF CONTENTS

1. INTRODUCTION

1.1 BACKGROUND

Grumman recently completed a feasibility study to determine what contribution atmospheric flight operations could make to the LEM Program (References 1 and 2). As a result of this work it was concluded that:

1. A preliminary design of a minimum modification of an all-rocket powered LEM (LTA-9) to be used for atmospheric test/flight experience should be conducted.

2. In addition to the currently planned LEM simulator program a free flight vehicle with LEM lunar handling characteristics is required to provide an acceptable level of crew training in the terminal descent portion of the landing maneuver. It was further concluded that an adaptation of the Lunar Landing Research Vehicle (LLRV) currently under development for the Flight Research Center (FRC) would represent a technically and economically attractive solution to this requirement.

3. A test version of the LLRV would permit early flight experience with the Reaction Control Subsystem and other LEM equipment vital to the descent and landing phase.

On April 12, 1963 a presentation of this work was made to MSC resulting in an MSC request that Grumman commence a preliminary design of an all-rocket LEM (LTA-9) for atmospheric test/flight experience (Reference 2). Scheduled completion date for this effort is 15 July 1963. In addition, Grumman was asked to investigate the use of LEM subsystems or components and design data and technology in the Flight Research Center's and Langley Research Center's (LRC) Lunar Landing Programs to increase their fidelity and applicability to the LEM program. A due date of 15 May 1963 was specified for the latter task to permit review of Grumman recommendations prior to the FRC LLRV design review scheduled for early June.

1.2 SCOPE

This report presents results of the work performed to date by Grumman to fulfill the Reference 2 requests as they pertain to the NASA lunar landing research programs.

Three general areas of application to the LEM development program have been investigated:

1. Early verification of LEM system analysis and design decisions.

2. Use of LRC and FRC vehicles for LEM hardware testing.

3. Use of the FRC LLRV for free flight experience in a LEM-type vehicle.

In view of the need to provide timely information pertinent to the LLRV design review, specific recommendations concerning performance, hardware installations, and payload capability have been directed toward this vehicle. The description

Code 26512 Eng-23A

of LEM handling qualities, physical characteristics, and performance capability which form the basis for these recommendations, is of course equally applicable to both programs.

1.3 REPORT CONTENT

This report contains three main sections, each of which deals with LLRV applications in one of the general areas cited above. Additional supporting material is presented in a series of appendices following the body of the report.

In Section 3, applicable LEM characteristics and systems are first defined and then compared with the corresponding FRC LLRV characteristics and systems. Recommendations are then made regarding changes to the LLRV which would increase the program's contribution in the area of early verification of LEM system analysis and design decisions.

In Section 4, LEM hardware subsystems which could be profitably tested on the FRC LLRV are defined. A physical description and indication of the weight of this equipment, instrumentation requirements and ground support equipment is provided so that installation problems on the LLRV can be evaluated. A development schedule indicating the availability of LEM equipment and the desired test period is also presented.

In Section 5, the requirements pertinent to obtaining maximum-fidelity LEM flight experience in the LLRV are discussed. Information based on current design studies concerning LEM crew capsule geometry, visibility provisions, and displays used during hover and landing is presented. Results of a preliminary performance investigation showing LLRV potential flight duration with representative training and test payloads are also shown, together with the assumed vehicle recovery provisions and configuration weight breakdowns on which the performance is based.

* * *

Preliminary investigations to date indicate that development of the LEM flight control system can be materially assisted by utilization of the LRC landing facility. However, in order to furnish LLRV recommendations by 15 May, completion of the LRC investigation was deferred and will be covered in more detail during the LTA-9 preliminary design phase. This effort will consider: use of the Langley lunar landing vehicle flight program for LEM design parameter investigations; installation of LEM flight control hardware, including the RCS, on the Langley vehicle; and possible operation of LTA-9 in the Langley facility.

Code 26512 Eng-23A

2. SUMMARY

GAEC has concluded as a result of the work presented here that, with suitable modifications, the LLRV program could contribute heavily to the LEM development in the area dealing with the terminal descent and landing phase.

Without introducing any changes which would delay the LLRV schedule (April 1964 delivery to FRC) the following LEM design concepts could be evaluated during early LLRV flights (Section 3).

1. Hovering and low speed flying qualities.
   Flight control powers and modes.
   LEM type performance.

2. Flight control and display configuration.

3. Crew accomodations.

4. General vehicle geometry.

The recommended modifications to achieve the above items are:

1. Use of the 16 LLRV control rockets in a configuration equivalent to LEM.
2. Install side arm type flight controllers (already anticipated in the LLRV program).
3. Implement LEM control jet logic equations.
4. Modify the LLRV attitude control system to include the LEM Emergency Minimum Impulse Bit mode.
5. Provide ground adjustable pilot seat tilt capability.

Present weight data indicate that sufficient lift capability is available in the LLRV to test LEM subsystem equipment vital to the lunar landing phase of the LEM mission. With the recommended modifications and incorporation of the LEM flight control system the LLRV will permit the first free flight manned test and evaluation of the LEM's bipropellant reaction control subsystem and its stabilization and control subsystem, and landing radar tests.

Investigation of the use of the LLRV as a flight test vehicle for LEM flight controls was based upon the Current LLRV Weight Statement, dated 5 April 1963. We have been informed that the LLRV configuration has since undergone extensive modifications. Before the test applicability of the LLRV to the LEM program can be firmly established, installation of the LEM equipment on the LLRV must be checked. Therefore, it is suggested that up-to-date configuration data including structural arrangement, weights and inertias be made available to Grumman.

Finally, the LLRV has sufficient lifting capacity so that with reasonable modifications (Section 5) it could provide the free flight training specified in the GAEC General Performance Criteria LEM Flight Crew Trainers (Reference 9).

These criteria include:

1. Provisions for both one and two-man operation.
2. LEM type flying qualities including descent engine simulation.
3. A close approximation of LEM flight controls and displays.
4. A conservative crew safety and vehicle recovery system.
5. Flight performance well beyond the nominal mission.

The recommendations advanced to achieve these items are:

1. Simulation of LEM attitude control configuration and modes as indicated on previous page.
2. Installation of a LEM type light-weight cabin envelope, controls, and flight displays.
3. Installation of a rapid deploying parachute.
4. Installation of a JATO final deceleration system.

The LLRV performance studies described in Section 5 shows that 6 minutes of one-man flight duration with a full rocket load (2 minutes of lift rocket time) and 350 pounds allotted for recovery provisions are attainable at FRC (Edwards Air Force Base) on a standard day. Two man operation under these conditions is not possible at FRC because of the turbojet engine performance penalties associated with the 2300 foot terrain height at this location. This limitation could be avoided, however, by reducing the duration of the lift rocket from two minutes to one minute.

The two-man training operation can also be carried at FRC by using the LLRV turbojet engine to provide the thrust usually supplied by the rocket engine, thus eliminating the weight of the lift rocket fuel. In this approach, some degradation in simulation of lift rocket response will occur, but should not seriously compromise training effectiveness.

Two-man extended rocket capability at the Ames Research Center, where the low field elevation results in improved turbojet thrust capability, was also explored. It was found that flight time is not sufficiently improved unless the uprated turbojet engine mentioned in Reference 4 is available.

Finally, a preliminary investigation has indicated that the lunar rocket fuel weight can be significantly reduced without compromising the rocket handling qualities at all by having the LLRV lift rockets provide only that part of the lunar rocket thrust over which the pilot has control on the LEM. Since the LEM descent engine cannot be throttled below a thrust/weight ratio of .5 and the LLRV lift rockets can provide a minimum thrust/weight ratio of .33, the difference (.17) could be supplied by the jet engine. If an operating minimum thrust/weight ratio of .83 was acceptable, half the original rocket thrust could be replaced by the jet engine thus saving about 300 pounds of fuel.

In view of the short design and fabrication time scheduled for the LLRV, it is expected that there will be limited opportunity to reflect modifications in the LEM design occurring after the LLRV design review date. The above recommendations are based primarily on published information regarding this vehicle and hence may not reflect the latest LLRV design. Therefore GAEC would welcome the opportunity to discuss the final LLRV design with appropriate NASA personnel.

3. LEM DESIGN DECISIONS SUITABLE FOR CHECK WITH EARLY LLRV FLIGHTS

In the terminal descent and landing phase of the mission, the LEM presents many of the design problems which are characteristic of VTOL aircraft operation on earth. However, the difference between the lunar environment and earth environment are so great that all aspects of the existing VTOL design technology must be reexamined to insure that all the consequences of lunar operation are adequately treated.

GAEC and others are doing this to the extent that analytical and ground based simulation techniques permit. However, it is expected that actual operational experience will significantly upgrade the final LEM design. The LLRV is designed conceptually with the flexibility to present many of the basic engineering problems inherent in a lunar landing, and will fly early enough so that the lessons learned on this vehicle can be integrated into the LEM. Hence, the Apollo program could benefit significantly, through early verification of LEM system design decisions provided the LLRV reflects the same engineering approach to the lunar landing problems as that used on the LEM.

The LEM systems which operate during terminal descent and landing must accommodate the requirements of other mission phases, and hence cannot in most cases be optimized for descent and landing. GAEC has reviewed the LEM system designs applicable to the terminal descent and landing to establish what LEM design approaches are now incorporated in the LLRV and what LEM design approaches could be incorporated in the LLRV without significant design modifications or schedule changes. This work is discussed in the following paragraphs.

3.1 HOVERING AND LOW SPEED FLYING QUALITIES

Lunar VTOL operation differs from earth operation in that the gravitational field is reduced and there is no atmosphere to provide angular or displacement damping or flight path stability. Reference 3 indicates that both these effects will be achieved on the LLRV, by controlling the thrust and tilt angle of the CF-700 turbo-fan engine located at the LLRV center of gravity.

Although this installation means that the LLRV must pair its descent rocket engines around the jet engine while the LEM incorporates a single lift rocket, the effects of this difference on the vehicle flying qualities will be small.

3.1.1 Attitude Control System

The LEM attitude control system comprises 16 rocket nozzles arrayed 45 degrees from the principal horizontal plane axes as shown in Figure 4.2, and it is recommended that an equivalent arrangement be installed on the LLRV. While LEM handling characteristics could be achieved with a number of arrangements, assuming all the rockets worked, the LEM arrangement provides a high degree of redundancy. Present work indicates that adequate flying qualities are retained with certain nozzles inoperative and it is necessary to verify these conclusions under realistic conditions. Hence, an LLRV nozzle array equivalent to the LEM together with LEM jet command logic equations would permit flight checks with selected nozzles out. The preliminary LEM logic equations are given in Reference 7.

The LEM control power variation during terminal descent and landing is given below:

Axis	Control Power	
Pitch	6.49 - 8.56	deg./sec.^{2}
Yaw	5.54 - 6.33	deg./sec.^{2}
Roll	6.77 - 7.32	deg./sec.^{2}

Since the LEM design will not be finalized for some time, it is recommended that the LLRV include a control power capacity range from 5 to 10 deg./sec.^{2} about each axis to cover possible LEM configuration changes.

The LEM attitude control system comprises the following manual modes:

1. Rate Command with Attitude Hold - In this mode the LEM tilts at a rate proportional to the control displacement and retains the attitude reached at the time the control is returned to neutral.

2. Attitude Command - In this mode the LEM tilts rapidly to a position proportional to the stick position. Since in hovering flight horizontal acceleration is proportional to tilt angle, this mode is equivalent to a horizontal acceleration command mode for translation maneuver purposes.

3. Emergency Direct On-Off - In this mode the pilot only has simple on-off control of the moment about each axis. It represents the most degraded control system condition. Moving base simulator investigations indicate that this mode can be flown acceptably with practice, but requires a high level of pilot concentration.

4. Emergency Minimum Impulse Bit - In this mode the pilot initiates a train of evenly timed pulses with any control displacement away from the neutral position. The qualitative effect is the same as in 3 above but the control sensitivity and fuel consumption is greatly reduced by the pulse effect.

3.1.2 Descent Engine Control System

In the terminal descent and landing phase of the mission the LEM descent engine thrust is controlled directly by a hand throttle. The response time of this system is so short that the pilot perceives a nearly instantaneous thrust response.

Although the LLRV hydrogen peroxide lift rockets will have a significantly slower response time than the LEM descent engine because of the difference in propellant characteristics, it is still anticipated that the inherent differences between the LEM and LLRV rocket system will be small enough so that installation of a LEM type throttle handle would permit a close approximation to the LEM control system. (See Section 5)

The LEM descent engine is gimballed so that the thrust vector can be tilted to pass through the center of gravity if it moves off the vertical axis. Since the thrust vector must be vertical in the hovering (and hence landing) condition the LEM attitude must be tilted relative to the horizon. On the LEM an automatic system drives the engine gimbal to accommodate center of gravity changes, hence the trim position for hover varies as much as 3 degrees in pitch and .5 degrees in roll (pilot axis) during the descent. Simulator investigations have indicated that this has a pronounced effect on pilot technique and it is recommended that the effect be provided on the LLRV. A time variable bias in the roll and pitch attitude reference system is suggested as a promising approach.

3.1.3 Translation Jet

For initial flights a conventional stick and rudder will be fitted to the LLRV, and the following discussion is based on this arrangement. The entire control system is of the fly-by-wire type, however, and the installation of other control configurations should present no problem.

Analysis has indicated that for small horizontal velocity changes less control rocket fuel is required if the jet is used directly for braking or accelerating. As a result the LEM control system permits firing of control jets to achieve horizontal acceleration directly. Accelerations of the order of .6 ft/sec.^{2} are obtainable on the LEM and similar values should be achievable on the LLRV. This control should not produce any appreciable tilting motions.

3.1.4 LLRV Attitude Control Capability

A review of Reference 3 indicates that the LLRV attitude control system has the capability to provide up to 0.8 rad./sec.^{2} control power about the pitch and roll axes, and 0.4 rad./sec.^{2} about the yaw axis (pilot axes). There are two direct control modes with no stability augmentation and two modes utilizing rate and/or attitude gyro signals.

In the first direct mode the stick and rudder are linked directly to the hydrogen peroxide values which meter fuel to the reaction jets, thus providing moment accelerations which are proportional to pilot control displacement. In the second direct mode, a potentiometer on the stick and rudder provides electrical signals to a pulse-width modulator activating on-off solenoid values which supply fuel to the control rockets. Again the control moments are proportional to control displacement, but at lower level than the first mode because of the pulsing effect. The pulse frequency is fixed at 1 pps, and pulse width can vary from .02 to .95 seconds according to the stick displacement.

The two stability augmentation modes also utilize the pulse modulator and solenoid valve arrangement, with signals generated by body tilt rate and attitude gyro signals as well as control displacement. In one mode the LLRV responds as a rate-command system in which body rate about each axis is proportional to control stick displacement. In the other mode both rate and attitude signals are used and the LLRV responds as an attitude command system in which body attitude about each axis is proportional to stick displacement.

In each of the stability augmentation modes the rate and attitude feed back signals can be varied by the pilot to obtain a wide range of system response performance.

Based on the above description of the LLRV control system configuration it appears that, with reasonable modification, the LEM control modes and system characteristics can be closely approximated with the LLRV . The main change would be a modification to the pulse width modulation system to simulate the LEM Emergency Minimum Impulse Bit Mode.

The block diagram of the LEM RCS configuration and the current range of system characteristics are presented in Appendix A as a guide for LLRV RCS system modification. It is expected that the jet thrust build-up obtained with the hydrogen peroxide jets will be appreciably slower than the LEM bi-propellant system. By proper design, however, this difference can be accommodated without seriously affecting the LEM simulation fidelity.

3.2 FLIGHT INSTRUMENTATION

The LEM flight control instrumentation layout and functions are presented in Figure 5.3 and Table 3-1. The LLRV flight instrumentation is presented in Figure 3.1 and Table 3-2. The LLRV display is from Reference 4 and the functions are from Reference 5. It is apparent that the flight instrumentation approach is generally the same for each vehicle and that the accuracies specified for the LLRV cover the ranges of interest for LEM.

There are some areas, however, where modifications to the LLRV design would improve its applicability to LEM. Although the LEM display does not now show it, serious thought is being given to including a pilot adjustable descent fuel direct analog readout of the pie or bar chart type. Since fuel in the tank corresponds roughly to the energy represented by attitude and airspeed in a conventional airplane, the direct analog display techniques used for these quantities in airplanes would be appropriate for descent engine fuel.

It should be noted in regard to the attitude display that the LEM mission requires large, long period angular excursions about the pilot pitch and yaw axes, and that certain mission phases require very accurate attitude control. Hence in configuring an attitude display using a ball presentation, care must be taken to avoid presenting a "pole" and thus degrading the presentation resolution. GAEC is currently investigating various all-axis attitude indication schemes, and it is suggested that FRC specification for an LLRV attitude display be coordinated with the LEM program to insure that the design features required by other aspects of the mission are adequately treated.

The vertical acceleration indicator on the LLRV display currently has its counterpart on the LEM propulsion system panel display. The method of presentation currently under consideration is shown in Figure 5.3.

TABLE 3.1

LEM BASIC FLIGHT INSTRUMENTATION

Key *	Item	Range	Accuracy
1	Altitude Rate, ft/sec	± 100	± 1 or 5%
2	Altitude, ft	0-100,000	± 5 or 1%
3	Roll Attitude, deg.	0-360	± 1
	Pitch Attitude, deg.	0-360	± 1
	Yaw Attitude, deg.	0-360	± 1
	Roll Attitude Error, deg. **	± 5 & ±.5	2%
	Pitch Attitude Error, deg. **	± 5 & ±.5	2%
	Yaw Attitude Error, deg. **	± 5 & ±.5	2%
	Roll Rate, deg/sec	±25 & ±5	2%
	Pitch Rate, deg/sec	±25 & ±5	2%
	Yaw Rate, deg/sec	±25 & ±5	2%
4	Range, ft	0-5000	±2 or 1%
5	Range Rate, ft/sec	±500	±1 or 1%
	Heading Velocity, ft/sec	±50	±1
	Drift Velocity, ft/sec	±50	±1

* See Figure 5-3

** Not applicable for terminal descent and landing.

TABLE 3.2

LLRV BASIC FLIGHT INSTRUMENTATION

Key	Item	Range	Accuracy
1	Altitude Rate, ft/sec	± 120	± 1
2.	Altitude, ft (fine)	0-2000	± 2
3.	Altitude, ft (coarse)	0-5000	± 25
4	Roll Attitude, deg	0-360	± 1
	Pitch Attitude, deg	0-360	± 1
	Yaw Attitude, deg	0-360	± 1
	Roll Rate, deg/sec	± 40	2 %
	Pitch Rate, deg/sec	± 40	2 %
	Yaw Rate, deg/sec	± 40	2 %
5	Heading Velocity, ft/sec	± 80	± 2.5
	Drift Velocity, ft/sec	± 80	± 2.5
6	Rocket Thrust Acceleration, g	± 1	± .05
7	Descent Fuel, ---	---	---

3.3 FLIGHT CONTROL AND WINDOW CONFIGURATION

The flight control and window configuration for the LEM have not been finalized to date & are currently the subject of extensive investigations. The following general characteristics seem well established, however. There will be a left-hand throttle control operating generally like an airplane throttle, and a right-hand attitude controller located in a vertical plane including the pilot's right shoulder. (See Figure 5.2)

The pilot will probably be seated with his back approximately parallel to the vertical axis to facilitate view of the ground during landing, and will be provided with a window configuration generally as shown in Figure 5.2. His eye position relative to the ground, the landing gear, and vehicle center of gravity are indicated in Figure 5.1. It is suggested that the incorporation of provisions to adjust the pilot seat-tilt be investigated in the LLRV design so that changes in the LEM configuration could be accomodated.

References 3 and 6 indicate that supporting structure to mount a flight control system of the general configuration discussed above will be provided on the LLRV, and that the pilot eye orientation relative to the landing gear and center of gravity will be generally similar to Figure 5.1. The degree of pilot forward tilt during the hover and landing phase is presently being studied in conjunction with current visibility investigations. It is expected that the pilot position will not be varied during terminal descent.

3.4 LANDING GEAR CONFIGURATION

The design criteria for the LEM landing gear are still under study, and it is expected that they will be modified from time to time as more detailed lunar surface information is obtained. In the Grumman Proposal, the LEM touchdown motion limitations were ± 5 ft/sec horizontal velocity, 10 ft/sec vertical velocity, and ± 5 degrees of total tilt.

The LLRV design criteria given in Reference 6 are as follows:

> For gear design purposes, it will be assumed that the jet engine thrust will be supporting 2/3 of the design weight of the vehicle at touchdown. The gear system shall be designed for a maximum rate of sink of 10 ft/per second for a vertical landing (no horizontal component) on level terrain with all legs contacting the ground simultaneously. On terrain so rough that all four legs do not contact the ground, and on slopes up to 15° or with a side drift velocity of 3 ft/per second, the gear shall perform satisfactorily for sink rates up to 6 ft/per second. Design limit loads shall be based on design gross take-off weight of the vehicle. Ultimate loads shall be limit loads times a factor of safety of 1.5.

In view of the extensive use proposed for the LLRV vehicles both for LEM hardware testing and crew training as well as applied research, it is suggested that design criteria equivalent LEM values be considered.

Code 26512 Eng-23A

3.5 TERMINAL DESCENT AND LANDING TECHNIQUES

The flight paths and pilot techniques, which will be most applicable for the LEM mission, are still under investigation, and it is anticipated that the results of early LLRV flights will contribute to this work. As a guide for design and planning purposes, however, a nominal flight path which is based on LEM mission planning is presented below. As a further guide maximum LEM performance is analyzed in Appendix B.

The key portion of the nominal trajectory is a relatively flat glide slope (perhaps 20 degrees) initiated a few hundred feet from the surface. During this glide, the sink rate should be comfortably below the landing gear limit sink speed, and the ground speed should be low enough so that it can be cancelled using reaction jets directly for braking with no resulting pitching motion. The LEM should be in the attitude-hold mode, or be kept in the vertical attitude by the pilot.

Such a glide affords the pilot the opportunity to scrutinize landing sites in the closest detail as he floats towards them at very low altitude, and only requires him to apply reaction jet braking to execute a deft landing anywhere along a strip several hundred feet long. A mild application of power will extend this strip still further, and abort possibilities remain good, since the LEM does not tilt, and the descent rates are low. Even some vision obscuration might be tolerated at the final touchdown in view of the constant attitude during the entire maneuver (attitude hold) and no need for last minute throttle changes.

Although the use of reaction jets directly for braking may seem extravagant compared to use of the RCS to tilt the LEM, the coupling between tilting and translating motions is such that to achieve velocity changes as low as 3 ft/sec quickly would involve 15 degree tilt angles and a period of 4 seconds using the RCS for moment control. These figures are particularly significant in view of the landing tolerances of $\pm$ 5 degrees and $\pm$ 10 ft/sec. The same RCS fuel used directly for braking could cancel 2 ft/sec in the same period with no pitch motion at all.

The final glide is preceded by one or more steeper glide segments with much higher descent rates and somewhat higher ground speeds. During each glide segment, the LEM remains vertical and the glide slope (or range) is adjusted with throttle.

The transition (flare) between glide segments is accomplished with a brief throttle pulse and pitch angle excursion using the RCS. Maximum pitch angles as high as 20 degrees may occur briefly during the middle of the maneuver, but for long periods the LEM glides in a steady erect position during which all system performance can stabilize and be monitored.

Descent fuel considerations require that very high descent rates be held until the final glide segment, and the LEM is capable of such maneuvers, having a vertical deceleration capacity of 10 ft/sec^2 (4200 ft/min reducing to 0 in 250 feet). While pilots have actually achieved such performance in special purpose aircraft, it requires considerable skill, practice, and familiarity with the landing site. Since neither the opportunity to practice (perform a build-up series) or the site familiarity will be available for the LEM mission; however, more moderate flares consistent with site scrutiny requirements and abort possibilities will be preferred.

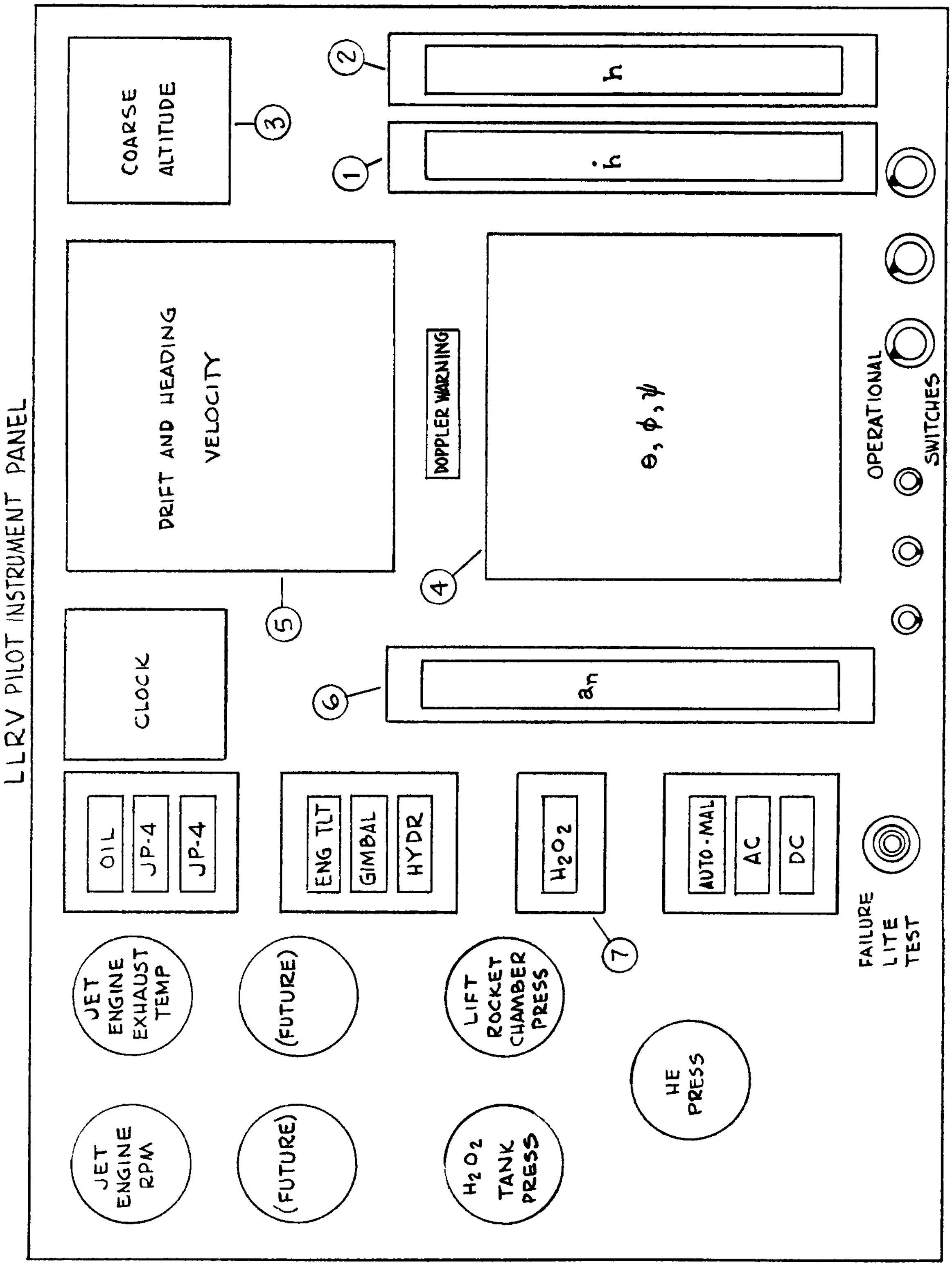
3.5.1 Estimate of Final Trajectory

The typical trajectory shown in Figure 3.2 reflects the above considerations. The initial descent rate of 1200 ft/minute and 51.5 degrees descent angle are well within conventional helicopter practice, and the moderate flare (970 pounds thrust increase and a brief pitch-up of 20 degrees) occurs at a descent rate and altitude combination such that an abort could be carried out if the descent engine faltered.

The final glide slope of 19.5 degrees is within steep descent airplane practice* and the rates and range to the surface are so low that a detailed appraisal of debris problems and surface irregularities can be made without the distractions associated with coordinated maneuvers. A landing can be made anywhere along a 330 foot strip without a major throttle change, and no large RCS fuel expenditure occurs until after a specific site has been selected and the problems associated with hovering directly over that site evaluated.

It will be noted that in the previous discussion no consideration was given to transition from a hover at 1000 feet altitude or to sideward flight. As explained previously, the strong coupling between tilting and translation motion and the relatively slow LEM response to tilt control makes any velocity change complicated if it is to be accomplished in a brief period. Hence, it is assumed that the pilot would prefer a 40-second initial glide during which range can be extended with a simple throttle adjustment, and a constant attitude shallow slow glide for 38 seconds directly over the landing area, rather than the distant view and the pitch maneuver involved in the 1000 foot hover. The fuel changes associated with this modification to the flight plan are negligible.

* The pilot likes to see the horizon and landing site simultaneously during the final approach.



LLRV PILOT INSTRUMENT PANEL
COARSE ALTITUDE
3
2
h
1
ḣ
DRIFT AND HEADING
VELOCITY
DOPPLER WARNING
θ, φ, ψ
OPERATIONAL SWITCHES
4
5
CLOCK
6
a_n
OIL
JP-4
JP-4
ENG TLT
GIMBAL
HYDR
H_2O_2
AUTO-MAL
AC
DC
FAILURE LITE TEST
7
JET ENGINE EXHAUST TEMP
(FUTURE)
LIFT ROCKET CHAMBER PRESS
HE PRESS
JET ENGINE RPM
(FUTURE)
H_2O_2 TANK PRESS

Fig. 3.1

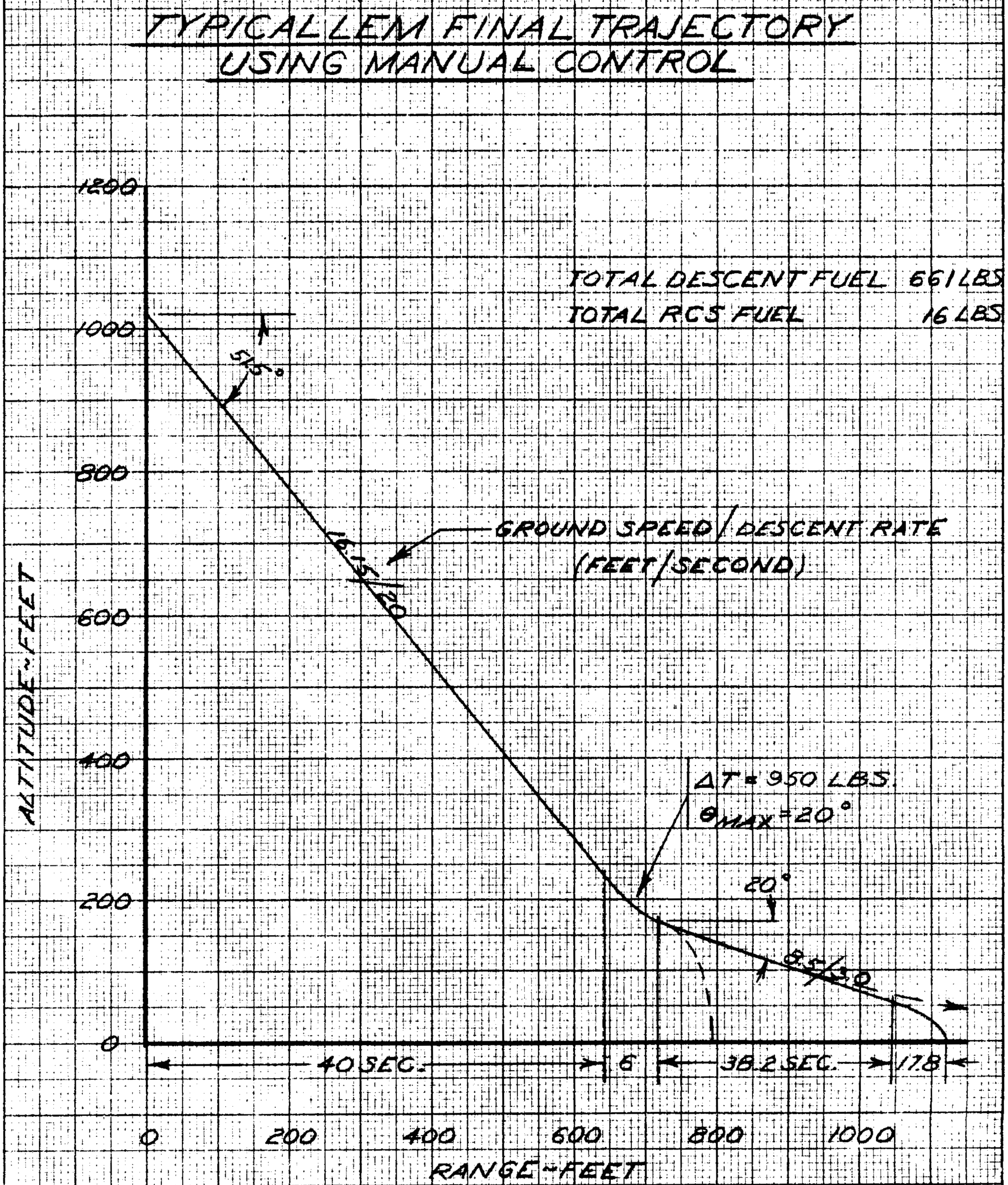

TYPICAL LEM FINAL TRAJECTORY
USING MANUAL CONTROL
TOTAL DESCENT FUEL 661 LBS
TOTAL RCS FUEL 16 LBS
51.5°
GROUND SPEED / DESCENT RATE
(FEET/SECOND)
16.5/20
ΔT = 950 LBS.
θMAX = 20°
20°
8.5/3.0
40 SEC.
6
38.2 SEC.
17.8
ALTITUDE~FEET
RANGE~FEET
0
200
400
600
800
1000
1200

Fig. 3.2

4. USE OF THE FRC LUNAR LANDING RESEARCH VEHICLE IN THE LEM DEVELOPMENT TEST PROGRAM

4.1 PURPOSE

The LLRV permits near-operational LEM sub-system usage and flight experience in an earth environment. It is the only manned free-flight vehicle, currently scheduled, in which LEM flight controls can be operated under closely simulated lunar velocity/attitude relations. Flight testing the man-managed integrated flight controls during the terminal descent and touchdown could, therefore, permit:

1. An evaluation of LEM equipment, and
2. An indication of the need for refinements of techniques and equipment in the most critical part of the LEM mission - lunar landing.

The LEM flight controls and landing radar will be tested and evaluated for both performance and response during terminal descent. The combined flight environmental exposures include inertial and elastic response, and touchdown loadings. The performance evaluation will be based upon analysis of ground-recorded airborne telemetry signals and qualitative flight crew opinion. The environmental response of LEM equipment will be monitored through post-flight inspection of equipment in conjunction with post-flight data analysis. All test flights will be continuously monitored through the use of air-to-ground telemetry and two-way radio link.

The LEM test configuration-LLRV gross weight of 3347 pounds results in brief flight times. It is estimated that for development testing, five minute turbojet powered flight durations will be required. Simulated lunar landing demonstrations or proof flights can be performed within two minutes. As indicated in Section 5. (See table 5.1), the technique for sustaining lunar simulated flight as well as the ambient atmospheric conditions at the test site influence the total flight time available. Therefore, before detailed flight test plans can be made, the following information must be confirmed:

1. The maximum usable CF700 thrust.
2. The technique for simulating lunar vehicle dynamics.
3. The location of the test site.

4.2 TEST CONFIGURATION

Investigation of the LEM flight controls aboard the LLRV will require modification of the basic vehicle. This modification includes installation of LEM Stabilization and Control Subsystem electronics and the LEM Reaction Control Subsystem (bipropellant, Aerozine 50 and N_2O_4). Provision for carrying and evaluating the LEM landing radar on a block of research/test flights is also desirable. The installation of operable LEM electronics also necessitates the environmental control of this equipment.

A general arrangement drawing of the proposed configuration is not shown since the basic LLRV configuration is being re-evaluated by FRC at the present time. However, the suggested planform incorporates the LEM landing gear arrangement, i.e., a forward, aft and athwart, four pad alighting gear. For reference, the pertinent LEM subsystem arrangements are included herein. A schematic representation of the LEM Stabilization and Control Subsystem is presented in Figure 4.1. The assemblies to be evaluated aboard the LLRV are indicated. In addition, the interfaces between S and C and LEM Guidance and Navigation and Descent Engine are evident. It is anticipated that any interfaces between the LEM assemblies to be installed and the LLRV can be "tied-off" by modification of onboard LLRV equipment based upon knowledge of LEM requirements. The dual tankage LEM Reaction Control Subsystem arrangement is shown in Figure 4.2. The Reaction Control Subsystem is shown, schematically, in Figure 4.3. For LLRV operations the propellant line tie-ins to the LEM Ascent Stage are not applicable.

The development scheduling of LEM hardware results in its earliest availability suitable for incorporation aboard the LLRV, in February/March 1965. It is anticipated that between first-flight date of the LLRV and the end of 1964 the structural integrity, the propulsion systems and basic flight control of the LLRV design will have been proven. The equipment common to both the basic LLRV and the LEM Test Configuration LLRV includes:

* Airframe Structure
* CF-700 Engine - Gimbal - Fuel Subsystem
* Lift Rocket Subsystem
* Flight Control Electronics Subsystem
* Ejection Seat
* Electrical Subsystem
* Research Instrumentation Subsystem.

The piloting experience and operational utilization of the LLRV during the FRC Phase I Research Program should provide the prerequisite confirmation and technique development of the above mentioned LLRV subsystems.

The LEM flight control equipment is to be evaluated with respect to:

* Vehicle control effectiveness
* Confirmation of control electronics logic, switching, stability and calibration
* Transient response and feedback - electrical and fluid flow.
* Effectiveness of reaction control pulse coding technique.
* Free flight structural dynamics - momentum carry-through and vibratory response.
* In-flight calibration check.

To take advantage of the LLRV's unique ability to provide the correct (lunar) vehicle attitude-velocity environment during terminal descent and touchdown it is necessary to arrange the thruster quads, fluid and electrical lines as similar to the LEM arrangement as is practical. This layout will reduce the variables that come under consideration in evaluating test results. However, a more important requirement is insuring that the test vehicle reflect the proper "LEM" dynamics, i.e., simulation of torque-to-inertia ratio. Increased translational response (when using LEM RCS in translational mode) may be inherent due to increased thrust to mass ratio, however, this should be tolerable.

Employing the LEM landing gear arrangement will extend the utility of the vehicle to include investigation of flight control during landing dynamics; i.e. the effect of landing translation and vehicle rotation after initial contact with the ground. Although the landing gear energy absorption characteristics, c.g. height above ground and tread may differ to some extent from the LEM, data to enable a "generic" or comparative study may be obtained. Under these conditions an evaluation of the response and interaction of LEM equipment and the (LLRV) vehicle can be made. In addition, LEM landing aid hardware such as penetrometer arrangements can be evaluated.

Specific LEM subsystem equipment to be installed on the LLRV for flight testing of the LEM control system is listed below:

Stabilization and Control Subsystem	Size (in.) L	W	H	Wt. (lbs)	Power Reqd. (watts)	Heat Dissip. (watts)
A. Attitude and Translation Control Assy.	19.5	10.125	6.5	20	35	35
B. Guidance Coupler Assembly	13.5	10.125	6.5	20	30	30
C. Rate Gyro Assembly	1.	7.5	6.5	9	115V, 45V 400∿ 3ϕ	36
D. Attitude Reference Assembly	20	10	6.9	25	115V, 80V 400∿ 3ϕ	64
E. Attitude Controller	6.25	3	3.5	5	-	-
F. Thrust Controller (reaction controls)	8	4.5	5.25	7	-	-

Reaction Control Subsystem	Weight (lbs)
A. Tanks and Supports	48.3
B. Pressurization System	27.3
C. Plumbing	57.5
D. Thruster Quads	80
E. Thruster Quad Supports	40

The cockpit controls and displays associated with the flight control tests aboard the LLRV are summarized in Table 4.1.

The cooling requirements for LEM Stabilization and Control Subsystem components and Landing Radar components will necessitate the use of equipment cold plates similar to those used in the LEM vehicle. The cold plates will be cooled by pre-cooled water (between 40-70°F) which will be pumped from an accumulator and dumped overboard after use. Such a system could be conditioned by a recirculating chilled water cart during preflight ground operations and checkout. The estimated weight for this environmental control system for a 30 minute operation is 30 pounds; of this total 20 pounds is coolant water.

TABLE 4.1 LEM Control and Displays For Installation Aboard the LLRV

Displays (Stabilization and Control)				Weight (lb)
A. Integrated Attitude Display				16.0
1. Attitude	Attitude Error	Rate		
a) Roll	d) Roll	g) Roll		
b) Pitch	e) Pitch	h) Pitch		
c) Yaw	f) Yaw	i) Yaw		
B. Altitude				1.5
C. Altitude Rate				1.5
D. Translation Velocity				
1. Forward				2.9
2. Lateral				
E. Clock				10.0
			Subtotal	31.9 lbs.

Displays (Reaction Control System)		
A. Dual Indicators (System A, System B)		
1. Helium Tank Pressure (A and B)		.8
2. Helium Tank Temperature (A and B)		.8
3. a) Fuel Quantity (A)		.8
b) Oxidizer Quantity (A)		
4. a) Fuel Quantity (B)		.8
b) Oxidizer Quantity (B)		
5. a) Fuel Pressure (A)		.8
b) Oxidizer Pressure (A)		
6. a) Fuel Pressure (B)		.8
b) Oxidizer Pressure (B)		
7. Thruster Fault Lights (8 required)		1.6
8. Pressure Regulator Malfunction Lights (4 required)		.8
9. Low Fuel Manifold Pressure Light (2 required)		.4
10. Low Oxidizer Manifold Pressure Light (2 required)		.4
	Subtotal	8.0 lbs.

Code 26512 Eng-23A

Table 4.1 LEM Control and Displays For Installation Aboard the LLRV (Cont'd).

Controls (Stabilization and Control)	Weight (lb)
A. Attitude Command	1.0
1. Roll	
2. Pitch	
3. Yaw	
B. Translation Control	1.0
1. Vertical	
2. Lateral	
3. Closing	
C. Select Switch	3.5
1. Attitude Control Auto-Manual (Attitude Hold-Attitude Command-Dir.-Minimum Impulse)	
2. Gyro (Primary Backup)	
Subtotal	5.5 lbs.

Controls (Reaction Control System)	
1. Pressurization Switch	.2
2. Regulator Shutoff	
a) System A Leg 1	.2
b) System A Leg 2	.2
c) System B Leg 1	.2
d) System B Leg 2	.2
3. Main Propellant Control (A)	1.0
4. Main Propellant Control (B)	1.0
5. Manifold Cross Tie Switch	.2
6. Eight Thruster Isolation Switches	1.6
Subtotal	4.8 lbs.

Total weight of LEM controls and displays for installation aboard the LLRV: 50.2 lbs.

Code 26512 Eng-23A

4.2.1 Support Equipment

Flight test operations with LEM equipment will require special test and handling equipment beyond that supporting the basic LLRV flight research program. Safety and special handling equipment and trained personnel are required because of the toxic, hypergolic propellant.

The type of support equipment associated with the LLRV-LEM test configuration include:

* Cart, High Pressure N_2 Supply
* Cart, High Pressure He Supply
* Cart, Flush and Purge Fuel and Oxidizer
* Cart, Fuel Storage and Transfer
* Cart, Oxidizer Storage and Transfer
* Cart, Propellant Vapor Disposal
* Cart, Vacuum, For Bladder Tank
* Cart, RCS, Subsystem Checkout
* Cart, Fuel Temperature Conditioning
* Cart, Oxidizer Temperature Conditioning
* Cart, Service-Freon (For pre-flight equipment conditioning)
* Cart, Service-Water
* Cart, Service Oxygen
* Bench, S and C Subsystem Test Equipment includes accelerometer, gyro, power distribution checkout and fault isolation.
* Console - Controls and Display maintenance, Test includes precision power supply AC and DC, circuit current and voltage adjustment panel, phase sensitive voltmeter, altitude and rate simulation panel, resistance limit bridge, precision signal generator, scopes and chart displays.

4.3 TEST INSTRUMENTATION AND DATA HANDLING

The primary data acquisition technique will be airborne telemetry linked to a ground based magnetic tape recording-playback system. In-flight data monitoring at the ground station will require real time strip chart displays and a radio link to the pilot. Post-flight data handling equipment should include an analog computing facility and provisions for oscilloscope displays, strip chart time histories, X-Y recordings, and frequency amplitude spectrums.

A firm instrumentation measurements list is beyond the scope of this recommendation and must await further LEM flight control development. A detailed listing will confirm the number of measurements required, range of measurements, frequency response and channel allocation regarding on/off status information, commutation rates and continuously recorded data. However, as an indication of the requirements a preliminary instrumentation measurements list is presented in Table 4.2. These items are additions to the basic vehicle velocity, acceleration, altitude and operational instrumentation that would normally be carried by the LLRV.

The recording of the continuous and time shared (commutated) measurement items, should be within the capabilities of IRIG standard PAM/FM/FM telemetry packages and tape recorders.

TABLE 4.2 PRELIMINARY INSTRUMENTATION MEASUREMENTS LIST

ITEM NO.	QTY.	PARAMETER	RANGE	FREQ. RESP.	ACCURACY
		Stabilization and Control			
1	3	Gyro Motor Supply Voltage	26v 400 ~	S/S *	5%
2	3	Gyro Torques Current	0-5v	S/S	1%
3	3	Accelerometer Torques Current	0-5v	S/S	1%
4	3	Gyro Bias and Align Volts	0-250mv	S/S	1%
5	3	Accelerometer Bias and Align Volts	0-250mv	S/S	1%
6	1	Roll Angle (from computer)	360°	10 cps	±.75°
7	1	Pitch Angle (from computer)	360°	20 cps	±.1°
8	1	Yaw Angle (from computer)	360°	10 cps	±.75°
9	1	Roll Rate (from Gyro Assy.)	±25°/sec.	1 cps	±.1°/sec.
10	1	Pitch Rate (from Gyro Assy.)	±25°/sec.	1 cps	±.1°/sec.
11	1	Yaw Rate (from Gyro Assy.)	±25°/sec.	1 cps	±.1°/sec.
12	1	Power Supply Output Current		S/S	1%
13	1	115v, 400 cycle P.S. output	115v	S/S	2%
14	1	15v, 5kc, P.S. Output	15v	S/S	2%
15	1	±50v DC P.S. Output	±50v	S/S	2%
16	1	Body Velocity Along Thrust Vector	0-100fps	S/S	±.5fps
17	1	Body Accel. Along Thrust Vector	0-2g	S/S	2%
18	1	Lateral Acceleration	0 to .5g	S/S	±1/2%
19	1	Pitch Rate Error	26v 800 ~	S/S	.1%
20	1	Roll Angle Error	26v 800 ~	S/S	.1%
21	1	Yaw Angle Error	26v 800 ~	S/S	.1%
22	1	Pitch Angle Error	26v 800 ~	S/S	.1%
23	1	Rate Gyro Temperature	0-180°F	S/S	2%
24	1	Power Supply Temperature	0-150°F	S/S	5%
25	9	Demod. Outputs (coarse)	±10v dc	5 cps	1%
26	9	Demod. Outputs (fine)	±1v	5 cps	1%
27	3	Dead Zone Outputs	±10v dc	5 cps	.1%
28	3	Limiter Zone Outputs	±10v dc	5 cps	.1%
29	9	Signals From Conditioning Stages (coarse)	±10v dc	5 cps	1%
30	9	Signals From Conditioning Stages (fine)	±10v dc	5 cps	1%
31	3	Rate Commands	±30°	2 cps	1%
32	3	Guidance	On-Off		
33	3	Attitude Hold	On-Off		
34	3	Attitude Command	On-Off		
35	3	Emergency Attitude	On-Off		
36	1	Pulse-Direct (Attitude)	On-Off		
37	1	Pulse-Direct (Translation)	On-Off		
38	6	Attitude Controller Detent Switches	On-Off		
39	1	Dead Zone Select	On-Off		
40	9	Rate Gyro Select	On-Off		
41	4	Logic Switch Positions	On-Off		
42	16	RCS Commands	On-Off	200 pulses/ sec. each	

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Table 4.2 PRELIMINARY INSTRUMENTATION MEASUREMENTS LIST (Continued)

ITEM NO.	QTY.	PARAMETER	RANGE	FREQ. RESP.	ACCURACY
		Reaction Control			
43	1	Helium Tank Pressure	0-5000 psia	300 cps	1%
44	1	Helium Tank Temperature	-20 to 120°F	S/S	2%
45	1	Helium Reg. Outlet Pressure	0-5000 psia	300 cps	1%
46	1	Helium Reg. Outlet Temp.	-20 to 120°F	S/S	2%
47	1	Helium Line Pressure	0-500 psia	300 cps	1%
48	1	Helium Line Temperature	-20 to 120°F	S/S	2%
49	1	Fuel Quantity Gage in Tank (Aerozine 50/50)	0 to 100 lb.	S/S	2%
50	1	Fuel Tank Pressure	0-300 psia	300 cps	1%
51	1	Fuel Tank Temperature	0 to 120°F	S/S	2%
52	1	Oxidizer Quantity Gage In Tank (N_2O_4)	0 to 160 lb.	S/S	2%
53	1	Oxidizer Tank Pressure	0-300 psia	300 cps	1%
54	1	Oxidizer Tank Temperature	0 to 120°F	S/S	2%
55	1	Fuel Main Line Pressure	0 to 300 psia	300 cps	1%
56	1	Fuel Main Line Temperature	0 to 120°F	S/S	2%
57	1	Fuel Main Line Flow Rate	.1 to 1 lb/sec.	10 cps	1%
58	1	Oxidizer Main Line Pressure	0 to 300 psia	300 cps	1%
59	1	Oxidizer Main Line Temp.	0 to 120°F	S/S	2%
60	1	Oxidizer Main Line Flow Rate	.2 to 1.6 lb/sec.	10 cps	1%
61	16	Chamber Pressure	0 to 120 psia	20-2000 cps	1%
62	32	Chamber Assy. Skin Temp.	0 to 3000°F	S/S	2%
63	16	Injector Housing Temp.	0 to 500°F	S/S	2%
64	4	Quad. Cluster Temperature	0 to 250°F	S/S	2%
65	32	Monitor Current at Each Thrust Control Valve	-	-	-
66	4	Oxidizer Line Pressure (Provisions)	0-300 psia	300 cps	1%
67	4	Oxidizer Line Temperature (Provisions)	0 to 120°F	S/S	2%
68	4	Fuel Line Pressure (Provisions)	0-330 psia	300 cps	1%
69	4	Fuel Line Temperature (Provisions)	0 to 120°F	S/S	2%
		Landing Radar			
70	1	Radar Altimeter Range	0v or -13v	2 cps	1%
71	1	Range Rate	0v or -13v	2 cps	1%
72	1	Horizontal Velocity	0v or -13v	2 cps	1%
73	1	Cross Track Velocity	0v or -13v	2 cps	1%
74	3	Beam-Mixer Transmittal Current	0v or -13v	S/S	1%
75	1	Output Power Monitor	-	S/S	-
76	1	Input Voltage	-	S/S	-
77	1	Timer Output	-	S/S	-

* S/S - Steady State

4.4 FLIGHT TEST PLAN AND SCHEDULING

The flight test program should commence in June of 1965. The test flights will be conducted using a one-man crew. Only one LLRV is scheduled for LEM hardware installation; furthermore, the performance estimates (Section 5.6) indicate that the LEM test arrangement results in one of the highest LLRV gross weight configurations. To assure attainment of the LEM test objectives a conservative buildup approach is anticipated.

4.4.1 Restrained Flight Tests

Initial tests will be tied-down telemetered runs to ascertain the compatibility of LLRV and LEM subsystems. Subsystem calibration and adjustment will be performed in conjunction with these tests. The environmental effects of the CF-700, the lift rockets and reaction control subsystems on the structure and fluid systems and electronic equipment will be determined. Approximately two months of tied-down test runs are estimated. Table 4.3 outlines these tied-down and tethered tests.

4.4.2 Free Flight Tests - Series 1

The first series of free flights will be short duration, low altitude, fully CF-700 powered tests. The primary objectives of this series are to:

* Familiarize the pilot with the vehicle

  a) cockpit arrangement
  b) vehicle dynamics
  c) touchdown dynamics

* Demonstrate altitude holding capability

* Investigate LEM open-loop acceleration control, i.e, Direct Mode.

* Investigate translational control with RCS using open-loop control.

* Investigate LEM Attitude Hold Mode, i.e., rate proportional to controller displacement - LEM pitch, roll and yaw axes.

* Investigate LEM Attitude Command Mode, i.e., attitude proportional to controller displacement - LEM pitch and yaw axes.

* Investigate time dependent and transient effects upon LEM equipment operation, i.e. fuel usage, sloshing, cooling, structural vibration, ground plane effects.

* Determine the capability of the LLRV-LEM test configuration to perform in an expanded flight envelope.

It is estimated that the first series of instrumented free flights could be completed and their results substantiated by data analysis during a 3 month time increment.

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4.4.3 Free Flight Tests - Series 2

During the Series 2 tests the LEM flight controls will be exercised by the pilot in simulated lunar terminal descents. At the completion of this series of runs the integrated man-LEM flight controls will have been demonstrated with regard to satisfactory performance during dynamic landing operations. During these tests it is expected that LEM equipment will be upgraded and refined as necessary. The bearing of these tests upon the basic LEM development program requires that flight results be made available to the LEM Project without delay.

These tests will require the use of the LLRV's capability for simulating the lunar gravitational environment. It is also expected that the LLRV hydrogen peroxide lift rockets will be utilized. These flight control tests will encompass longer durations and wider profiles than the Series 1 runs. Vehicle response data will be obtained in the different LEM flight control modes and the effects of mode switching in flight will be confirmed.

Landing dynamics under simulated lunar gravitation will be investigated after the ability of the gimballed jet engine to remain vertical in the presence of rapid touchdown attitude changes is established. The initial landing dynamics flights may be performed using a tethering arrangement that would permit angular and translational freedom to limited degrees. Depending upon the tethered test results, unrestrained flight demonstration of unusual attitude recovery techniques may be made.

A block of tests will be devoted to confirmation of the LEM landing radar. In particular, the use of the radar as a dependable terminal descent aid will be investigated. Other LEM landing aids such as penetrometers will also be evaluated during these flights.

Table 4.4 outlines the scope of the anticipated LLRV-LEM Test Configuration free flight tests. These activities are scheduled for a six-month period ending in December of 1965.

4.4.4 Development Scheduling

Modification of a basic LLRV to the LEM Test configuration is contingent upon availability of qualified LEM equipment in February and March 1965. The LEM equipment scheduling is such that the flying qualities and structural integrity of the basic LLRV can be demonstrated during the initial phase of flight research program, reference 8, planned by FRC.

A preliminary development schedule for the LEM test configuration LLRV is presented in Figure 4.4. Layup of an LLRV for modification and installation of the required LEM equipment is scheduled from January through mid May of 1965. Throughout the layup period, LLRV flight research and development will continue on No. 1 LLRV at Edwards.

Subsequent to delivery and checkout of the modified LLRV at the test site the buildup flight test program will commence. The LEM test objectives should be accomplished by the end of February 1966.

The Grumman, LEM flight crew training plan, reference 9 indicates usage of an LLRV by two LEM astronaut crews during March 1966. The LEM test configuration LLRV, No. 2 vehicle in Figure 4.4 will be available to the astronauts to complement the crew training LLRV configuration, should this be required.

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TABLE 4.3

LLRV FLIGHT TEST PLAN OUTLINE - RESTRAINED FLIGHT

1965
Mid June-July (or 1 month)

TIED DOWN TESTS

- Equipment Functional Test
  - GSE Inputs Applied
  - Communications Check
  - Environmental Control Check
  - Control Jet Sequence
  - Control Parameter Adjustment
  - Hot RCS Check
- Jet Engine Operating, Functional Test
  - Environmental Control Check
  - Control Jet Sequence (Hot RCS)
  - Structural Response (Primary and Secondary)
  - Control Parameter Adjustment
- Cockpit Check
  - Display Arrangement
  - Lighting
- Environmental Control
  - Toxic and Noxious Gas Control Check
  - System Flow Checks
- Ground Support Operational Confirmation
  - Pre-Flight/Post Flight - Electronics
  - Pre-Flight/Post Flight - Mechanics
  - Hypergolic Propellant Handling
  - Data Handling

1965
Mid July-August (1 1/2 month)

SHORT TETHER TESTS

- Jet Thrust Vectoring
  - Altitude Hold with RCS Operative
  - Attitude Hold with RCS Operative
- Stabilization and Control
  - Rate Gyros, Accelerometers
  - Attitude and Translational Control
- Handling Qualities
  - Control Effectiveness
  - Trim Changes vs. Fuel Quantity
  - Touchdown Stability and Dynamics
  - Degraded RCS Operation
- Structural and Acoustic Vibrations
  - Equipment Mounting and Structural Interaction
  - Equipment Operation
- Vehicle Loadings
  - Thermal
  - Aerodynamic Effects due to CF-700
  - Thrust
  - Inertial

Table 4.4

LLRV FLIGHT TEST PLAN OUTLINE - FREE FLIGHT

1965 - 1966

1965					
Sept	Oct	Nov	Dec	Jan	Feb

FLIGHT TESTS - SERIES I

- Flight Controls and Handling Qualities
  - Dynamic Response
    - Attitude Hold Mode
    - Attitude Command Mode
    - Emergency Attitude Mode
  - Control Parameter Adjustment
- Stabilization and Control Electronics
  - Terminal Stabilization and Control
  - Attitude Hold
  - Attitude Hold
  - Translational Control
  - Rate Gyros and Accelerometers
- Structural and Acoustic Vibration
  - Equipment Mounting
  - Structural Interaction
  - Equipment Operation

FLIGHT TESTS - SERIES II

- Maneuvers
  - Vehicle Response to Gross Attitude Changes (Max $\approx 30°$)
  - Stabilization and Control System Degradation
  - Translations and Surface Reconnaissance
  - LEM Abort Procedures Check
  - Velocity Profiles
- Landing Radar
  - Instrument Landing Procedures

SHORT TETHERED TESTS - POSSIBLE FREE FLIGHT DEMONSTRATION

- Landing Dynamics
  - Unusual Attitude Recovery
  - Attitude Hold Mode Failure
  - Attitude Command Mode Failure
  - Touchdown Dynamics Following Initial Contact

[redacted]

LEM STABILIZATION & CONTROL SUBSYSTEM

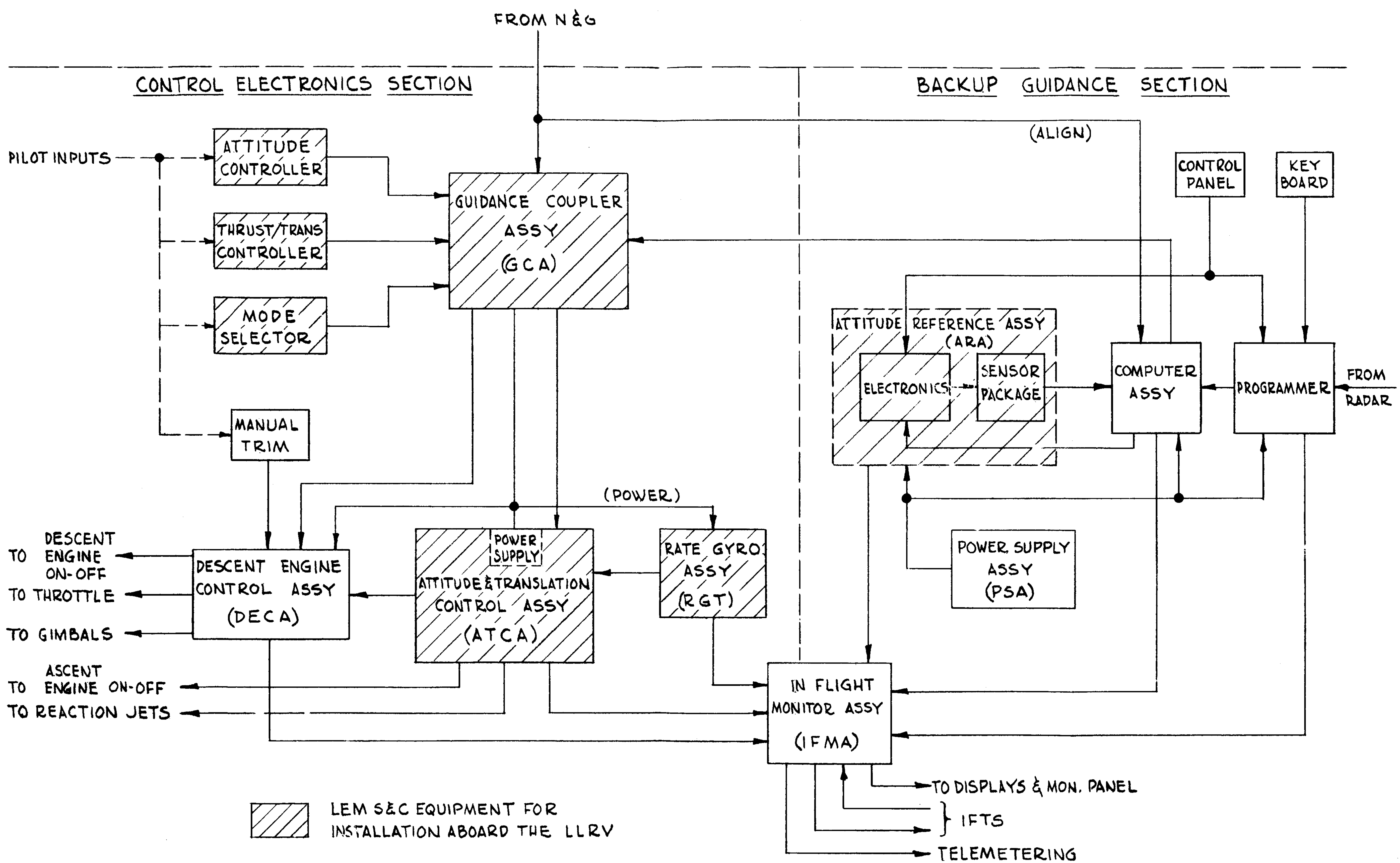

Fig. 4.1

[redacted]

[redacted]

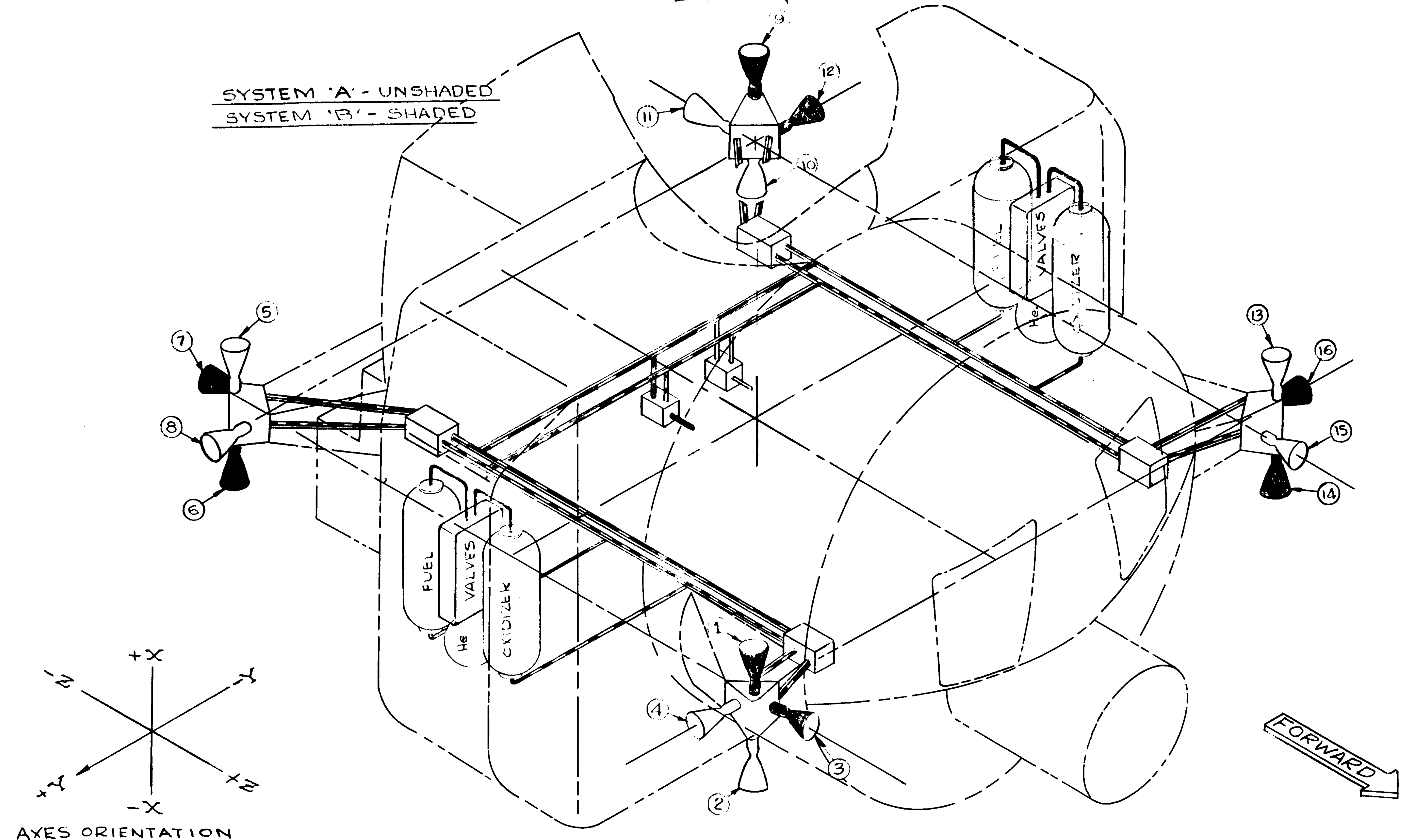

Figure 4.2
Reaction Control Subsystem

[redacted]

[redacted]

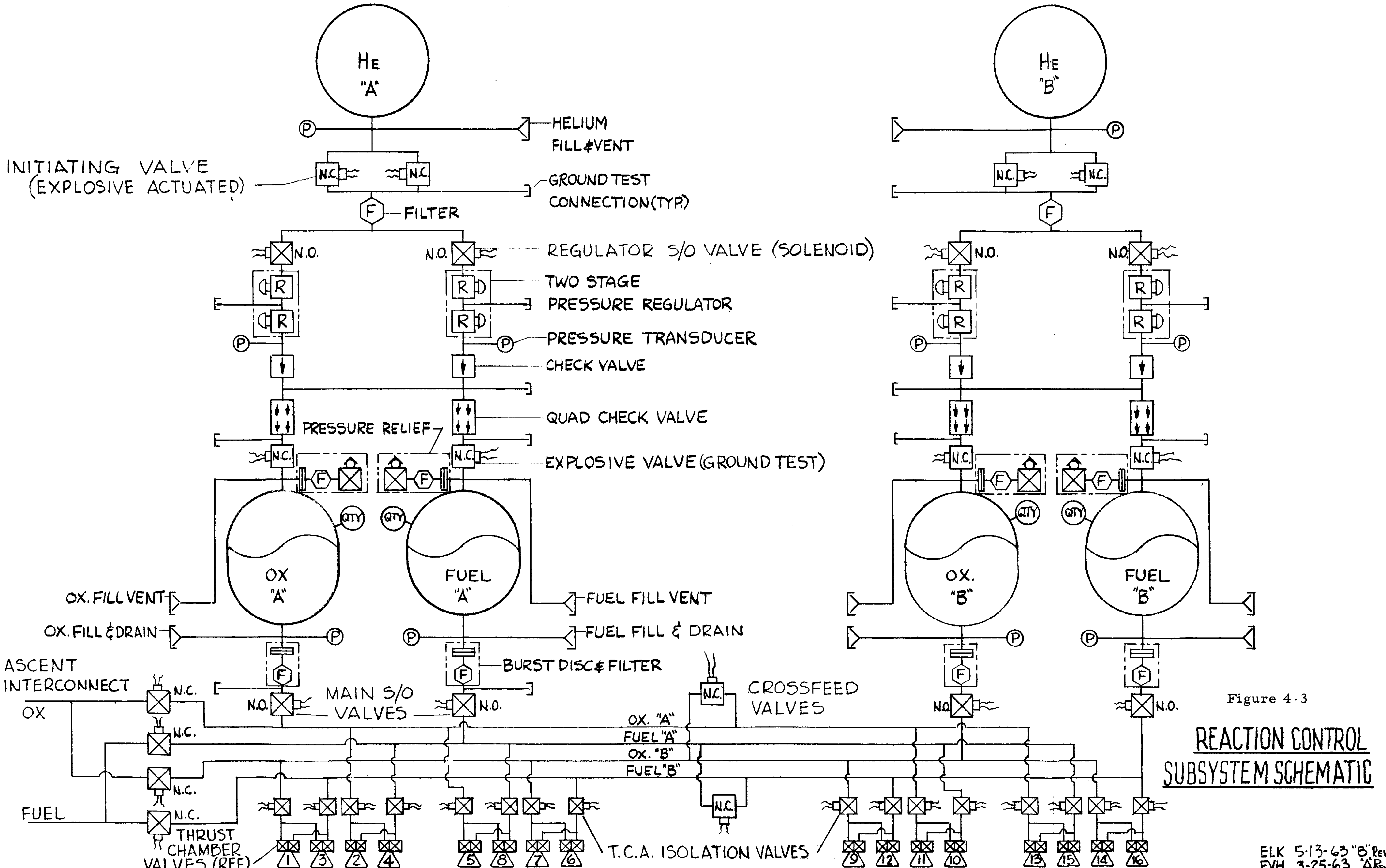

Figure 4·3

REACTION CONTROL SUBSYSTEM SCHEMATIC

[redacted]

PRELIMINARY - LEM TEST CONFIGURATION - LLRV DEVELOPMENT PROGRAM

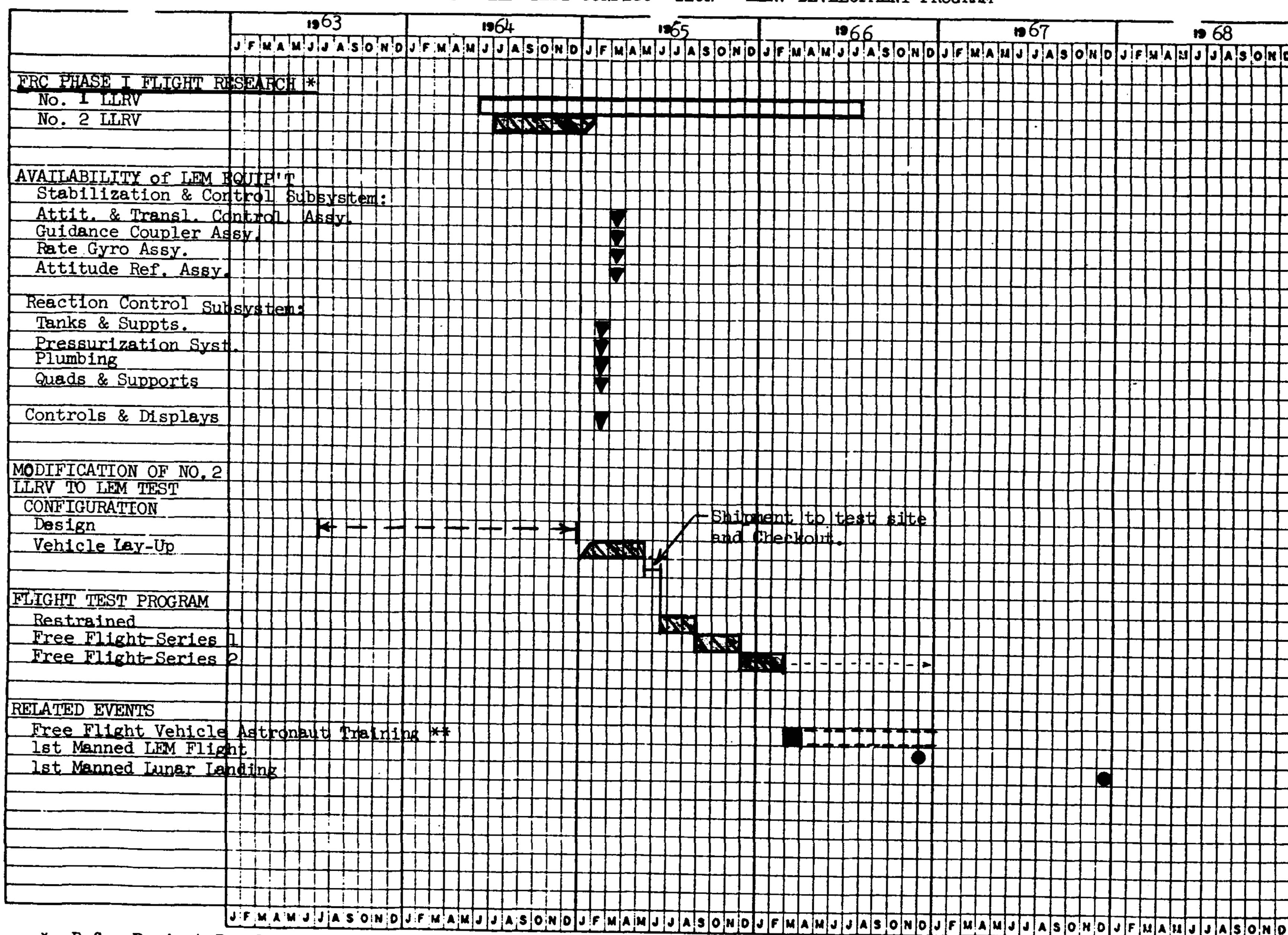

Figure 4.4

* Ref: Project Development Plan, Free Flight Lunar Landing and Take-Off Research Vehicle, FRC, 12/62

** Ref: GAEC Report No. LPL 480-1 - Training Plan for the LEM, Vol. 1 - 5/15/63

5. TRAINING REQUIREMENTS FOR FREE FLIGHT VEHICLE

5.1 INTRODUCTION

This section outlines the characteristics required of the Flight Research Center (FRC) Lunar Landing Research Vehicle (LLRV) to provide for astronaut flight experience in the terminal descent and touchdown phase of the LEM mission. Three areas of discussion are covered. First, the basic capabilities required of the vehicle are described. These requirements are based on the contents of GAEC Report No. LED-440-1, "General Performance Criteria, LEM Flight Crew Trainers", dated 15 May 1963.

Next the physical characteristics of the crew compartment pertinent to lunar landing training are reviewed. These are based on current LEM configuration data and include size, weight and location of displays and controls used during the terminal descent plus crew position and visibility geometry.

Results of a preliminary design study of LLRV performance for the LEM training mission are next presented. Vehicle weights used in the performance studies are derived from the Bell Aerosystems Current Weight Statement dated 5 April 1963. (See Appendix E). Performance data in terms of flight duration versus payload is presented for operation at the NASA Flight Research Center (FRC) and at the NASA Ames facility. In the performance studies the use of a "jet-engine only" mode of operation with compensation for the earth's gravity and aerodynamic effects are shown to permit two-man flights of useful duration with the necessary payload.

5.2 SUMMARY OF REQUIRED CHARACTERISTICS

The atmospheric free flight training vehicle will be used to provide the astronauts with flight experience in the performance of the terminal descent maneuver. The basic capabilities required of such a vehicle are summarized below:

1. Simulation of LEM response to attitude control and rocket throttle commands.
2. Simulation of the lunar gravitational and vacuum environment by automatic compensation for 5/6 of the earth's gravitational force and cancellation of atmospheric aerodynamic forces and moments.
3. Simulation of the LEM crew capsule geometry. Significant items are (a) pilot placement with respect to displays and controls, and (b) window size and location with respect to pilot.

4. Capability for operation in all three LEM manual attitude control modes: (a) Attitude Hold Mode, (b) Attitude Command Mode, and (c) Direct (Emergency) Mode. (These modes are further described in Section 3 of this report.)
5. Similarity to LEM pilot orientation with respect to the landing gear and pilot height above the ground at touchdown should be provided within the constraints of the LLRV design criteria.
6. Performance envelope of the LLRV should include as a minimum the nominal LEM landing maneuver, from 1000 feet to touchdown. Capability for training in flight profiles exceeding the nominal in altitude, range, and velocity, as described in Appendix B, should be incorporated to the greatest possible extent.
7. Provisions for two-man operation to permit instructor/student type operation and simulated LEM two-man landings, in addition to one-man capability.

5.3 LEM SYSTEM SIMULATION

Four LEM subsystems are of primary importance during the landing maneuver and must consequently be installed or simulated in the free flight training vehicle. These are:

(a) The Stabilization and Control (S & C) Subsystem.
(b) The Reaction Control Subsystem (RCS).
(c) Descent Propulsion Subsystem.
(d) Instrumentation Subsystem (portion of).

Based on our present knowledge of the FRC LLRV, it is believed that the first three subsystems mentioned above can be simulated with some modification by the stabilization system, peroxide attitude control system, and the peroxide rocket engine system currently planned for the LLRV. It is desirable, however, to replace existing LLRV flight instruments with pertinent elements of the actual LEM display panel during the training operation. As an aid toward obtaining the best possible simulation of LEM characteristics, additional information on the aforementioned subsystems is furnished below.

5.3.1 LEM Stabilization and Control Subsystem

5.3.1.1 Modes

Three possible manual attitude control modes can be used during the terminal descent phase of the lunar landing. For adequate training, it is essential that the LLRV be capable of simulating the correct LEM response to control inputs for each mode. Control modes are discussed in Section 3 and briefly repeated below.

The Attitude Command Mode will be the primary mode for translation during hover and descent. In this mode the pilot commands LEM pitch and yaw* attitude proportional to controller displacement with fore and aft, and lateral motion of the (vertically oriented) stick respectively. With the stick in the neutral position the vehicle returns to the vertical. This mode does not control motion about the roll (vertical) axis.

The Attitude Hold Mode will be used during hover and descent for movement about the LEM roll axis and for back-up of the Attitude Command Mode in the event of a malfunction. In this mode the pilot commands a vehicle attitude rate proportional controller displacement. With the controller returned to neutral the vehicle will hold the last commanded attitude.

The Emergency Attitude Mode provides back-up control in the event of failure of the alternate modes. Here the astronaut commands vehicle rotational acceleration on an individual axis basis through open loop control.

5.3.1.2 Control Power

The control power range experienced by LEM during the descent from hover to touchdown is summarized below:

	Hover	Touchdown
Pitch	$6.49^{\circ}/sec^2$	$8.60^{\circ}/sec^2$
Roll (LLRV Yaw)	$6.77^{\circ}/sec^2$	$7.32^{\circ}/sec^2$
Yaw (LLRV Pitch)	$5.54^{\circ}/sec^2$	$6.33^{\circ}/sec^2$

These values are based upon current LEM configuration data and are subject to change until the detail design of the vehicle is completed. It is expected that final values will remain within the range of 5 to $10^{\circ}/sec^2$. Capability for simulating average values of control power during hover and descent will be required of the free flight vehicle in the training application. The reader is referred to Section 3 for more detailed discussion of LEM attitude control system characteristics.

* LEM pitch corresponds to LLRV pitch. LEM roll and yaw correspond to LLRV yaw and roll respectively. (See figure 5.1)

5.3.2 Reaction Control Subsystem

The LEM reaction control system is comprised of four quadrants of four nozzles each positioned symmetrically about the vehicle c.g. along centerlines which bisect the cruciform landing gear at angles of 45 degrees (see Fig. 5.1). Vehicle attitude is controlled by introduction of pure couples about the vehicle pitch, yaw and roll axes. A translational mode of RCS operation is also afforded through unidirectional operation of the thrusters in symmetrical pairs. The thrusters may be operated at a thrust level of 100 pounds or alternately in a minimum impulse mode for precise attitude control. Further details on operation and physical characteristics of this system have been presented in Sections 3 and 4 respectively.

During the landing maneuver the RCS will be used in the 100 pound thrust mode with the control powers cited above. It is expected that the translational mode of operation will also be used in the Z (fore and aft) direction for terminal braking just prior to touchdown. The following preliminary values are applicable for simulation of handling characteristics during the braking maneuver:

(a) Braking thrust-to-mass ratio at touchdown $= \dfrac{200}{\dfrac{10,471}{32.2}} = .593 \text{ ft/sec}^2$.

(b) Induced moment (thrusters not in plane of c.g.) = 4600 in-lbs.

The LEM S & C subsystem will automatically compensate for this moment.

The difference in response time between the LLRV peroxide attitude rockets (60 ms) and the LEM rockets (12 ms) will not significantly effect the LEM control simulation.

5.3.3 Descent Propulsion Subsystem

LEM descent propulsion is provided by a throttleable rocket engine with an overall thrust ratio of 10 = 1. Thrust level used during terminal descent ranges from 1050 to 3500 pounds. The engine is gimballed about 2 axes to maintain vehicle trim in the presence of small c.g. shifts. A possible method of simulating this effect has been cited in Section 3.

The pertinent LEM propulsion system characteristics to be closely approximated are listed below:

1. Capability for vertical acceleration varies from 4.2 ft/sec^2 at hover to 5 ft/sec^2 at touchdown.

Code 26512 Eng-23A

2. LEM engine thrust response to step input equivalent time constant will be less than .3 seconds for thrust change of 1000 pounds with rocket operating at 1050 to 3500 pound level. (Engine Specification value)
3. Resolution (threshold) of LEM thrust control system = 2% of nominal thrust.

5.3.4 Crew Capsule Displays and Controls

Displays required in the LLRV for training in the terminal descent maneuver are those providing attitude, altitude, velocity and engine status information. Pilot's flight controls will consist of a mode switch for selection of the desired attitude control mode; a 3-axis attitude controller mounted on or adjacent to the crew's right-hand arm rest; and a thrust controller for translation and main engine thrust control located on or adjacent to the left-hand arm rest. A summary table of hover and landing instrumentation and controls used in the LEM is presented below.

ITEM	WEIGHT
A. Displays	
1. Integrated Attitude Display - 3 axis attitude display - pitch, roll and yaw rate	16.0 pounds
2. Attitude & Altitude Rate	2.9
3. Heading & Lateral Velocity (Z & Y)	2.9
4. ΔV Remaining	3.75*
5. Helium Tank Pressure	.53*
6. Regulator Malfunction Indicators (lights)	.03*
B. Controls	
1. Attitude Control Mode Selector Switch	3.5
2. Attitude Controller (3-Axis)	5.0
3. Thrust Controller	7.0

* LLRV lift rocket instrumentation should include similar presentations.

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Based upon the LLRV-LEM display comparison presented in Section 3, the LEM display requirements pertinent to hover and landing training will be satisfied by the instruments currently programmed for the LLRV. From the standpoint of training fidelity, it will be desirable to either simulate the appearance of (or use) the actual LEM components, located correctly with respect to the pilot. As an aid to future planning in the LLRV program, the external and internal geometry of the LEM crew capsule, including placement of displays, is briefly reviewed in the following paragraphs. It is expected that revisions to capsule interior geometry will occur based on the results of current design studies. However, the material presented reflects the present LEM design concept.

5.3.5 Crew Capsule Geometry

5.3.5.1 External Geometry

The LEM general arrangement is shown in Figure 5.1. The crew capsule is aligned with the forward leg of the cruciform landing gear at a height which places the pilot's eye 16 ft. 9 in. above the ground line at vehicle touchdown. Primary visibility during landing is afforded by two forward-facing windows located symmetrically with respect to the vehicle X-Z plane.

5.3.5.2 Internal Geometry

Preliminary capsule internal geometry is shown in Figure 5.2. The windows provide vertical and horizontal visibility of 53.5 and 60 degrees respectively when the pilot's eye is in the normal position (15 degree forward tilt). Additional visibility is obtained by further tilting of the astronaut's seat in a forward direction, and also by small horizontal windows placed in front of and below the astronaut. Final window configuration and the position to be assumed by the astronaut during terminal descent are the subject of current design studies.

Display and control consoles are located above and between the windows. An additional inclined console, cut out in the vicinity of the horizontal windows, is placed forward of and below the pilot. The panels for the reaction control subsystem (test only), the stabilization and control mode select panel (test and training), and the radar altimeter power and mode switches (test only), are situated between the windows on the vertical and upper panels.

An enlarged view of preliminary arrangements for these panels is shown in Figure 5.3. Also shown is a preliminary arrangement for the main propulsion instrument panel, presented for information only. It is not expected that this panel would be utilized for either the test or training application of the LLRV. However, it is desirable from the standpoint of crew training for the LLRV rocket engine instrument presentation (He and H_2O_2 pressure, thrust to weight ratio, fuel or ΔV remaining, etc.) to approximate that of LEM in appearance and location relative to the pilot.

An alternate crew capsule configuration also under consideration is shown in Figure 5.4. Primary change is in the capsule forward face, where the horizontal step has been replaced with a sloping lower face. In this arrangement the lower window has been enlarged, and upper window size may be asymmetrical about the X-Z plane.

It is expected that seat forward tilt would be a constant 15 degrees for all flight operations, assuming a vertical position only when LEM is resting on the lunar surface.

5.4 MONITORING OF TRAINING FLIGHTS

Monitoring of crew performance may be achieved by direct flight instructor participation, ground monitoring/tracking, and use of telemetry or airborne magnetic tape recording. The weight lifting capability of the final LLRV configuration will determine the method or methods selected. A preliminary investigation of performance (see Section 5.6) based on the LLRV weight statement of 5 April 1963 indicates a capability for two-man operation under selected conditions and operational modes.

For complete monitoring and evaluation of pilot performance the use of a telemetry or airborne tape recording is required. Parameters useful for evaluation and subsequent critique include:

* Altitude and Altitude rate.
* Lateral and Heading rates ($\dot{Y}$ and $\dot{Z}$).
* Pilot attitude and throttle control inputs.
* Attitude and Attitude rates.
* Engine thrust.
* Δ V or propellant consumption.
* Attitude control mode selection.
* Acceleration of pilot and vehicle c.g.

Use of the earth environment compensation system employed in the LLRV introduces an additional monitoring requirement. Provision should be made to discriminate between vehicle responses simulating LEM lunar dynamics and responses differing from LEM dynamics as a result of malperformance of the earth environment compensation system. This information should be either immediately displayed during flight or available for debriefing after completion of a training exercise.

5.5 CONTROL SYSTEM MANAGEMENT

The pilot tasks associated with control system management during the terminal descent-to-touchdown phase of the lunar landing are summarized below to indicate the basic procedures envisioned.

(a) Hold vehicle at hover (if applicable) - the pilot observes velocity and altitude indicators and employs control as required to keep V_H & V_V at zero.

(b) Roll vehicle to observe landing site - the pilot will command roll with the attitude controller.

(c) Determine slope of lunar terrain - pilot may be required to maneuver vehicle to obtain terrain slope information.

(d) Deploy landing aids - the type of landing aids to be used and the method of deployment have not been determined. Included among the aids under consideration are pyrotechnic flares, landing lights and penetrometers.

(e) Maneuver vehicle to touchdown point - typical maneuvers are discussed in Section 3.5.

(f) Experience Lunar Touchdown - the tasks involved in touchdown involve close control of horizontal and vertical velocities. Procedures involved in touchdown will include:

1. Monitor "Y" & "Z" rate indicators.
2. Monitor altitude and altitude rate indicators.
3. Monitor attitude indicator.
4. Pitch or yaw vehicle with reaction controller to thrust opposing excess velocity direction. Bring horizontal velocity to less than 5 ft/sec. and erect vehicle. Small adjustments in translation may be made using the thrust controller (RCS in translational mode).
5. Using thrust controller bring vertical velocity (sink rate) to 10 ft/sec. or less.
6. Experience touchdown - shut down and check all applicable systems.

5.6 SYSTEM PERFORMANCE

5.6.1 Introduction

In order to assess LLRV capability to provide a useful flight duration while supporting the required training and test (Section 4.) payloads, a preliminary investigation of potential vehicle performance in terms of payload versus flight duration has been carried out. The performance study considered:

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1. Three payload groupings
   a) Training; one man crew
   b) Training; two man crew
   c) Testing; one man crew plus LEM equipment described in Section 4.

2. Three operational environments
   a) Edwards AFB (2300 ft. terrain ht.); warm day
   b) Edwards AFB; standard day
   c) Ames Research Center (40 ft. terrain ht.); standard day

3. Four modes of operation
   a) Jet engine only*
   b) Jet engine plus 2 minutes of jet and rocket supporting 5/6 and 1/6 of the vehicle weight respectively
   c) Identical to (b) but with helicopter assisted launch
   d) Jet engine plus 2 minutes of reduced-thrust rocket operation. (See Appendix C)

4. Two versions of the CF-700-2B jet engine
   a) Standard; S.L. std. nominal thrust = 4300 lb.
   b) Uprated; S.L. std. nominal thrust = 4600 lb.

The results of the preliminary performance investigation are summarized on Table 5.1. Safety assumptions, vehicle weight derivations, and performance calculations and working curves are discussed in Sections 5.6.2, .3, and .4, respectively. Highlights of Table 5.1 are discussed below.

Briefly, the LLRV can provide one-man training flight durations in excess of five minutes for all modes of operation and environments considered with the exception of warm day conditions at Edwards AFB for missions involving two minute of rocket operation (mode 3b). This restriction can be lifted if a) jet engine nominal thrust is increased to 4600 pounds, or b) mode 3d above is used.

Two-man training flight durations in excess of five minutes are possible for all locations and environments considered if jet-only operation is used (T nom = 4300 pounds). Missions of greater than five minutes duration involving two minutes of rocket time are possible only at Ames with a) an up-rated engine, b) use of a helicopter launch, or c) use of mode 3d above. Two-man flights using rocket operation are possible at Edwards of rocket firing duration is reduced. For instance, a two man, $4\frac{1}{2}$ minute flight is possible at Edwards on a standard day with rocket propellant for one minute of firing plus the 265 pound reserve assumed for safety provisions (See Appendix D).

* Information from Bell Aerosystems indicates that earth environment compensation can be accomplished in this mode. This is a requirement if beneficial 2-man training is to be attained during jet only operation.

VEHICLE OPERATING MODE		One Man Test Version	
		T=4300 lbs	T=4600 lbs
a) Jet only operation	Edwar	3.5 min.	8.3 min.
	Edwar	8.1	13.2
	Ames	12.6	17.6
b) Jet plus 2 min. rocket operation	Edwar	0.6	1.2
	Edwar	1.2	1.8
	Ames	1.7	5.0
c) Helicopter Assisted Launch (jet plus 2 min. rocket)	Edwar	0.8	2.0
	Edwar	1.5	2.1
	Ames	2.0	6.9
d) Jet plus 2 min. rocket with jet simulating rocket thrust below throttleable rocket range.	Edwar	0.9	-
	Edwar	2.1	-
	Ames	7 0	-

* Nominal Thrust at Sea Level, Stanc

Test flights which carry the full complement of LEM equipment described in Section 4 plus one pilot and the reserve rocket propellant mentioned above can attain better than 5 minutes of flight with jet engine-only operation at Ames or Edwards on a standard day. An up-rated engine provides better than 5 minutes of duration for all sites and environments considered. Attainment of total flight times greater than 5 minutes which include two minutes of rocket operation can be attained at Ames (or any site having similar terrain height) using an up-rated engine or the nominal engine in conjunction with mode 3(d). Again, trade-offs are possible to improve over-all flight duration at the expense of rocket engine duration.

The preliminary investigation summarized above and discussed in more detail in following sections demonstrates that reasonable flight durations are attainable with the LLRV for both the test and training applications. The performance figures point up the advantage of operation in the jet-only mode for two-man or test missions, although the pilot will not actually be flying a rocket. Improved duration of rocket-powered flight for heavy payloads can be attained by a) operation at Ames, b) use of an up-rated engine, and c) implementation of the mode of operation suggested in Appendix C.

5.6.2 <u>Vehicle Recovery Provisions.</u>

Crew safety in the LLRV is provided by zero altitude, zero velocity ejection seats. In the event of failure of either the jet engine or the lift rocket engines, it is intended that the other system shall be a suitable means by which the pilot can effect a safe landing (Ref. 6).

The lift rocket system consists of 8 rockets which generate 500 pounds of thrust each. Two rockets, operating as a pair, are throttleable and provide approximately two lunar g of retarding thrust. The remaining six rockets are for emergency recovery and are only grossly throttleable (Ref. 4). In addition, a small drogue chute with a forcible deployment device is provided to stabilize the vehicle in the vent of attitude control failure and to reduce the terminal velocity in the event of a jet engine failure at an altitude too high for recovery by means of the lift rockets. (Ref. 6).

In considering vehicle operation in the jet-only mode, it was necessary to establish the amount of rocket propellant required to effect a safe landing in the event of a jet engine failure. A brief investigation was undertaken which considered:

a) Drogue chute weight and terminal velocity vs. distance required for deceleration to 10 feet/sec. at available thrust-to-weight ratios. (T max. = 4000 lbs.)

b) Rocket propellant required to effect a safe landing at altitudes below the vehicle parachute "dead-man" altitude of 100 to 150 feet.

Results of the investigation are detailed in Appendix D. Here the need for relatively low terminal velocities in the parachute system is demonstrated for the case in which the lift rocket system, with a relatively moderate thrust to weight ratio, is used to decelerate the vehicle to touchdown velocity. For a thrust-to-weight ratio of 1.1, 370 feet of altitude is required to decelerate the vehicle from a terminal velocity (with chute deployed) of 100 feet per second to a touchdown velocity of 10 feet per second. Less than 50 feet is required for deceleration from a terminal velocity of 40 feet per second.

Total weight of the system, including the parachute, is approximately 350 pounds. This penalty was added to the vehicle weight for both jet-only and (conservatively) jet plus rocket engine operation as a fixed weight payload and is included in the vehicle weight derivations presented in Section 5.6.3.

5.6.3 Weight Derivations for Training and Test Applications of the LLRV.

The weights used in the vehicle performance studies described in Section 5.6.4 were derived using the LLRV Current Weight Statement of 5 April 1963 as a base. A revised weight empty was first established by replacing and/or adding equipment as appropriate under each of the twelve major items comprising the LLRV empty weight. To this weight was added that of the crew plus all other non-expandable payload items for a given mission.

The resulting value, essentially gross take-off weight less jet fuel and rocket propellant, is the "end-of-flight weight" which forms the ordinate of the performance plots presented in Figures 5.5, 5.6, 5.7 and 5.8.

In all cases, the following ground rules were observed:

1) A 350 pound allowance for a vehicle recovery parachute plus emergency rocket propellant was included in the fixed weight.

2) The 200 pound payload allowance for FRC instrumentation and special electronics payload was maintained.

Weights derivations based on this procedure are shown for a) training configurations with one and two-man crews and b) a test configuration carrying the payload established in Section 4. (See Tables 5.2, 5.3 and 5.4 respectively.)

5.6.4 Estimated LLRV Performance

The LLRV performance was determined to establish payload - duration characteristics for the previously described operational modes. The performance calculations are based on the engine data presented in Reference 3, and use ground rules consistent with those listed in Reference 6. The engine used on the LLRV is a vertical version of the CF700-2B which is currently rated at a minimum guaranteed thrust of 4200 lbs. (S.L. Std. day) or a nominal thrust of 4300 lbs.

TABLE 5.2

WEIGHT DERIVATION; ONE-MAN TRAINING CONFIGURATION

Item (App. E)	Weight Change	Remarks
1. Structure	0	-
2. Alighting Gear	0	-
3. Controls - Manual		
Flight	-12	Replace LLRV stick & rudder pedals with LEM flight controller.
Engine	0	-
Rockets	-2	Replace with LEM thrust controller.
4. Controls - Automatic	0	-
5. Power Plant	0	-
6. Rocket System	0	-
7. Instruments	+12	Replace LLRV flight & navigation instruments with LEM flight displays.
8. Hydraulic & Pneumatics	0	-
9. Electrical	0	-
10. Communications	0	-
11. Furnishings	0	-
12. Auxiliary Gear	-50	Remove original drogue chute allowance.

Total Wt. Change:	-52
Original Wt. Empty	2169
Revised Wt. Empty	2117 Pounds
Recovery Provisions	350
LLRV Payload	200
Oil & Press. He.	12
Crew (1 man)	200
Total Wt. (less Jet and Rocket Fuel):	2879 Pounds

TABLE 5.3

WEIGHT DERIVATION: TWO-MAN TRAINING CONFIGURATION

Item	Weight Change	Remarks
1. Structure	+17	Add'l turnover structure.
	+12	Add'l windshield weight.
	+17	Add'l seat support structure.
2. Alighting Gear	0	-
3. Controls - Manual	-2	Replace LLRV stick and rudder pedals with LEM attitude and thrust controllers (2 ea.)
4. Controls - LLRV auto.	0	-
5. Power Plant	0	-
6. Rocket System	0	-
7. Instruments	+12	Replace LLRV flt. instrument with one set of LEM flt. instrument.
8. Hydraulics and Pneumatics	0	-
9. Electrical	0	-
10. Communications	0	-
11. Furnishings	+109	Add second crew station ejection seat, chute, harness and oxygen.
12. Auxiliary Gear	-50	Remove existing drogue chute allowance.

Total Wt. change:	+115
Original Wt. empty:	2169
Revised Wt. empty:	2284
Recovery provisions:	350
LLRV Payload:	200
Oil & Press. He.:	12
Crew (2 men):	400
Total Wt. (less jet and rocket fuel):	3246

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TABLE 5.4

WEIGHT DERIVATION: TEST CONFIGURATION

Item	Weight Change	Remarks
1. Structure	0	-
2. Landing Gear	0	-
3. Controls - Manual	-14	Replace LLRV flight controls and rocket throttle with LEM flight and thrust controllers.
4. Controls - Auto.		
LLRV	0	-
LEM	+76	LEM S & C electronics.
	+30	LEM landing radar.
	+25	Additional equipment support structure.
5. Power Plant	0	-
6. Rocket system		
LLRV	0	-
LEM	+253.1	Add LEM RCS incl. tanks, plumbing and supports.
7. Instruments	21.2	Replace LLRV flight and nav. instrument with LEM flight & RCS instrument.
8. Hydraulic and Pneumatics	0	-
9. Electrical	0	-
10. Communications	0	-
11. Furnishings	0	-
12. Auxiliary Gear	-50	Remove original drogue chute allowance.
	341.3	
Original wt. empty:	2169.0	
Revised wt. empty:	2510.3	
Recovery Provisions:	350.0	
LLRV Payload:	170.0	Removed FRC landing radar (Replaced with LEM Ldg. Radar in (4) above).
Oil & Press. He.:	12.0	
Add'l Environmental Cont.	30.0	(For LEM S & C Electronics)
LEM RCS Propellant	75.0	
Crew (one man):	200.0	
Total wt. (less jet &/or rocket fuel):	3347.3	

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For unassisted takeoff, a vehicle weight/thrust ratio of .93 was used based on the nominal thrust or the equivalent weight/thrust ratio of .955 based on minimum guaranteed thrust. These values were specified by Reference 6 and represent a a T/W = 1.05 for acceleration during the initial climb. For helicopter assisted takeoffs, it is assumed that the vehicle will be released at an altitude 1000 feet above the terrain. Under this condition, a vehicle weight/thrust ratio of 1.00 based on the minimum thrust rating was chosen.

Four types of mission were investigated to determine their effect on flight duration.

1. A zero rocket-time mission where the entire flight profile is performed without use of the rockets. Sufficient rocket propellant is carried, however, to decelerate the vehicle from recovery chute terminal velocity or permit a safe let down from a "dead man" altitude of 150 feet in the event of jet engine failure.

2. A two minute rocket mission where the vehicle is supported by the jet engine until the final two minutes of the flight. At this point, the jet engine is throttled back to a thrust equal to 5/6 of the vehicle weight and the rocket engines are started to simulate the LEM thrust-to-mass ratio.

3. A two minute rocket mission similar to the one above except that the vehicle is transported to the hover altitude by a helicopter. With this approach climb fuel is saved and a jet thrust-to-weight ratio of one may be used with a resultant increase in payload capability.

4. A jet plus rocket mission similar to mission (2) above which takes advantage of the approach suggested in Appendix C. Here only the portion of total rocket thrust over which the pilot exercises control is provided by the jet engine. This approach affords a substantial reduction in the hydrogen peroxide required for two minutes of lift rocket operation.

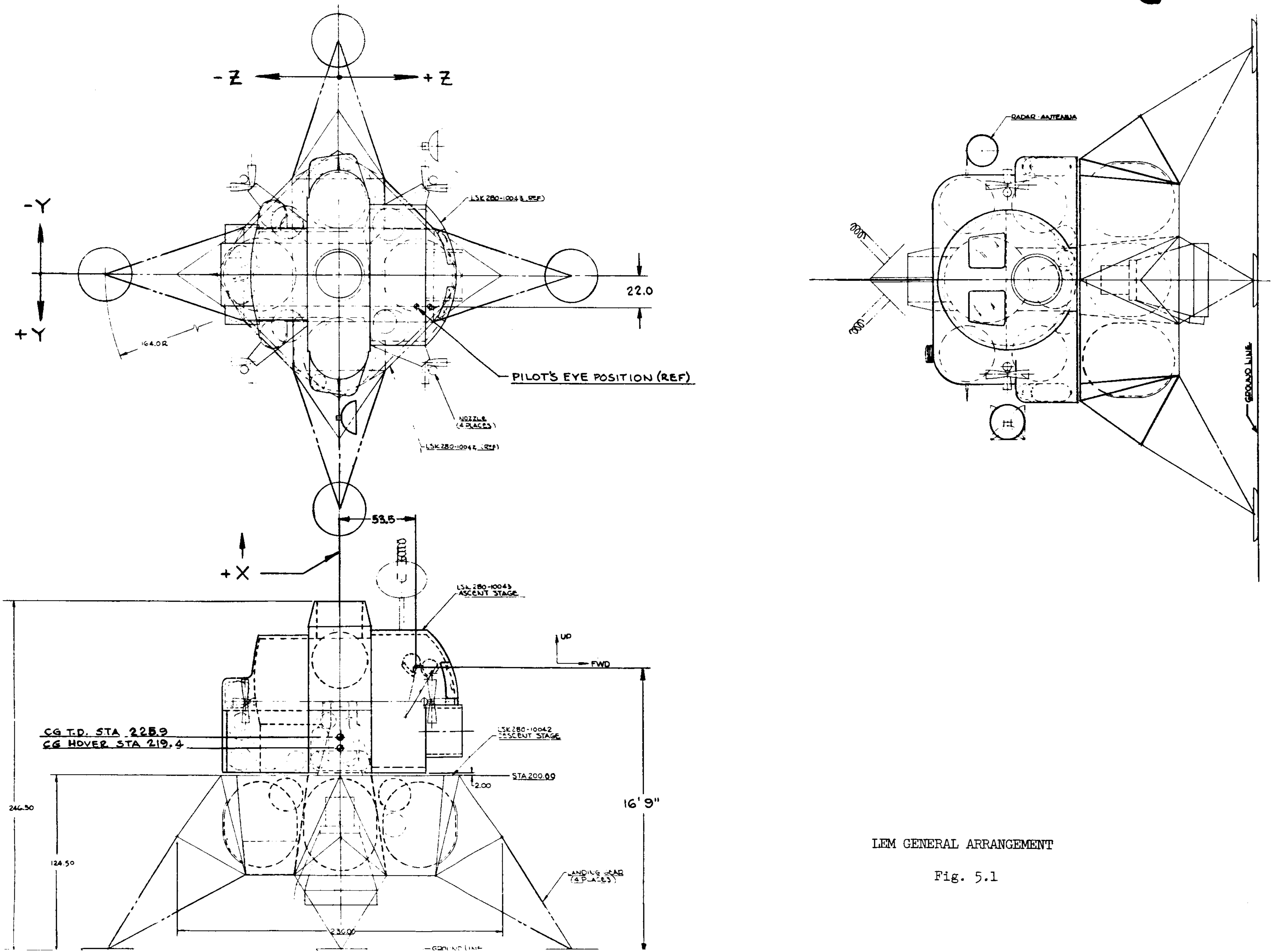
The ambient conditions considered are:

Edwards	- 2300 feet	Standard Day
Edwards	- 2300 feet	Warm Day (std. + 27^{o}F)
Ames	- sea level	Standard Day

Jet engine fuel flow obtained from Reference 3 was increased by 5% for service allowance. Rocket propellant was computed based on a specific impulse of 122 sec. Control propellant (hydrogen peroxide) consumption is estimated to be 10 pounds per minute based on a safety factor of 2. Propellant consumption and the safety factor are taken from Reference 3.

Performance was also calculated using an uprated version of the present engine. Reference 4 indicated that an increase of approximately 300 pounds in thrust may be possible at the expense of engine life based on similar J-85 trade-offs. The resulting nominal thrust of 4600 pounds provides a substantial increase in vehicle payload capability.

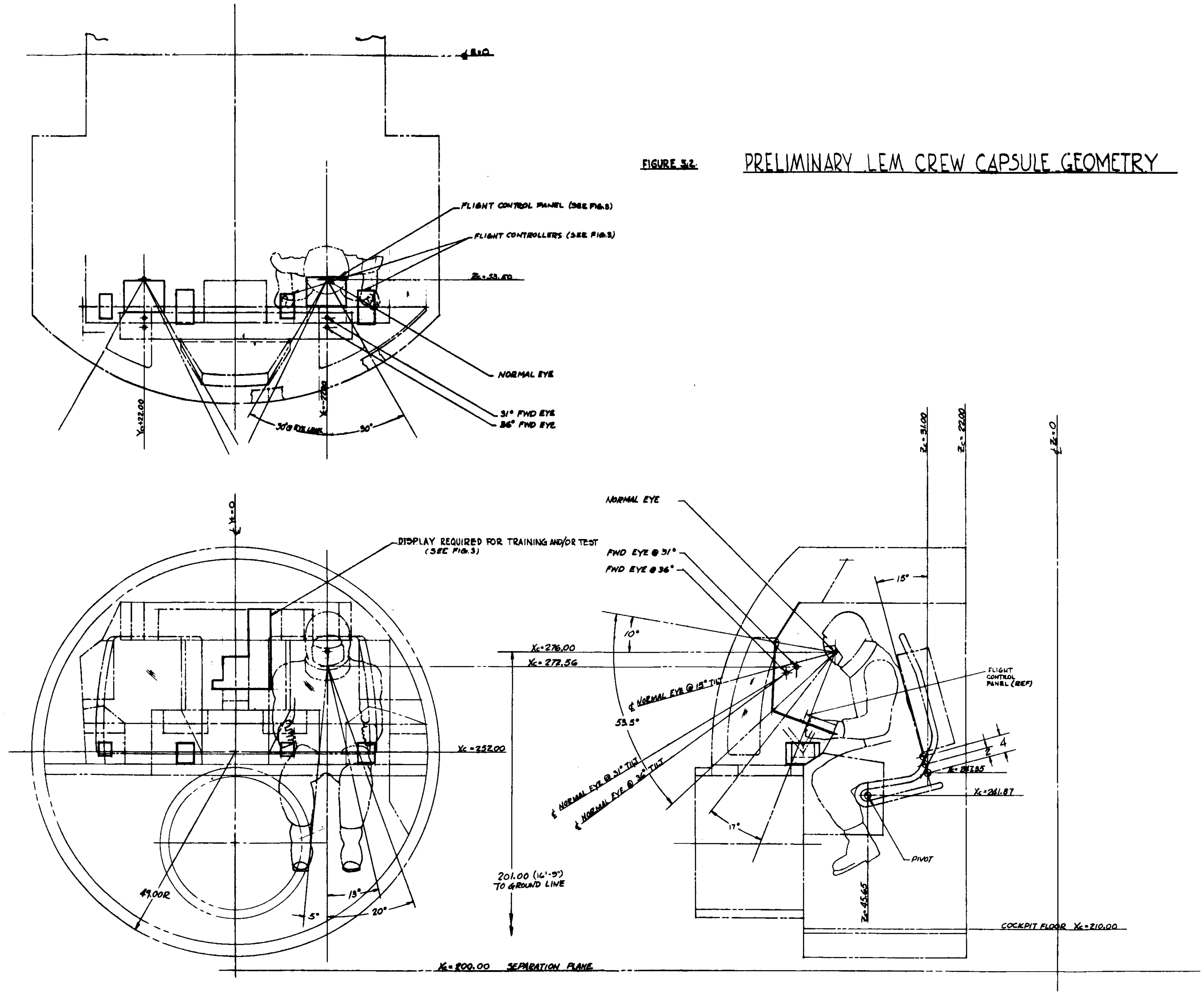
The resulting performance at Edwards for warm and standard day conditions, and Ames for a standard day is shown in Figures 5.5 through 5.8 for the operational modes considered. Performance is presented in terms of vehicle dry weight (take-off gross weight less jet fuel and rocket propellant) versus flight duration in minutes. Results for the dry weights associated with the training and test versions of the LLRV have been tabulated in Table 5.1



LEM GENERAL ARRANGEMENT

Fig. 5.1

FIGURE 3.2. PRELIMINARY LEM CREW CAPSULE GEOMETRY



[redacted]

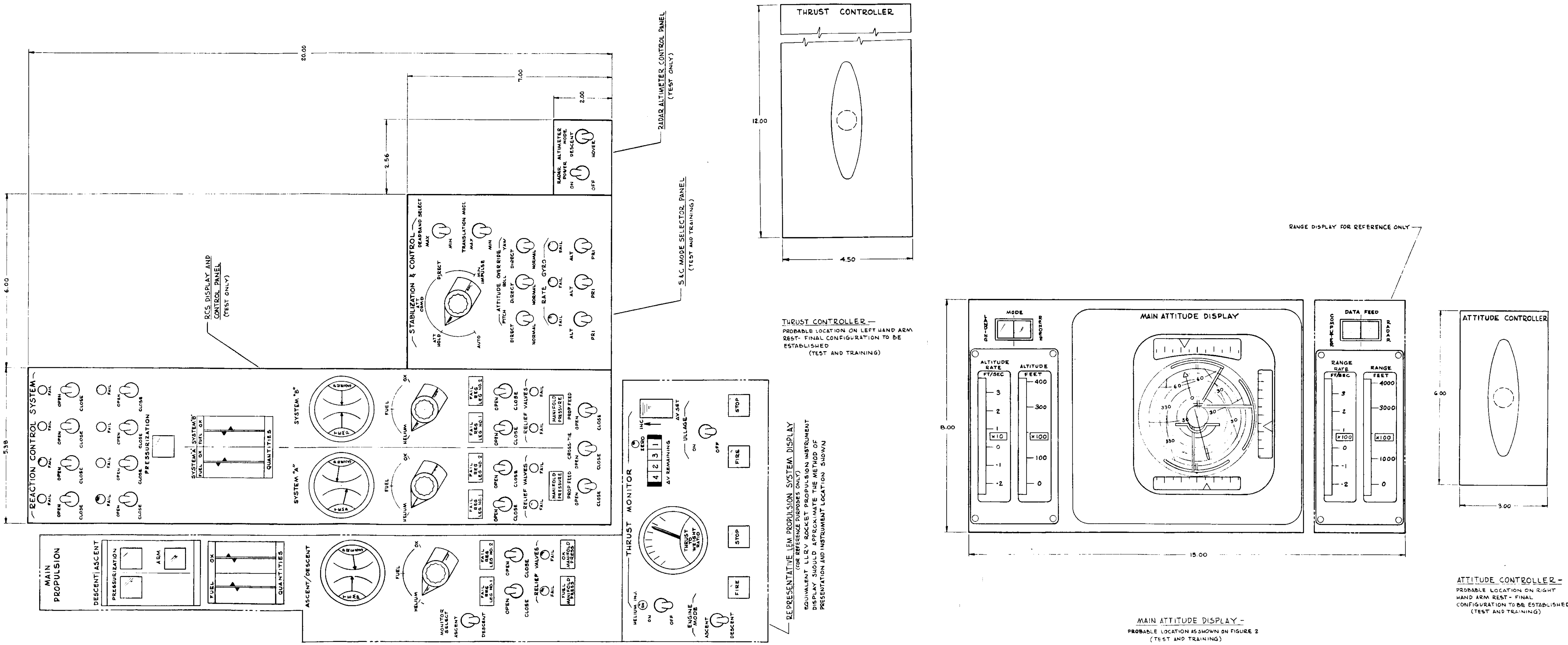
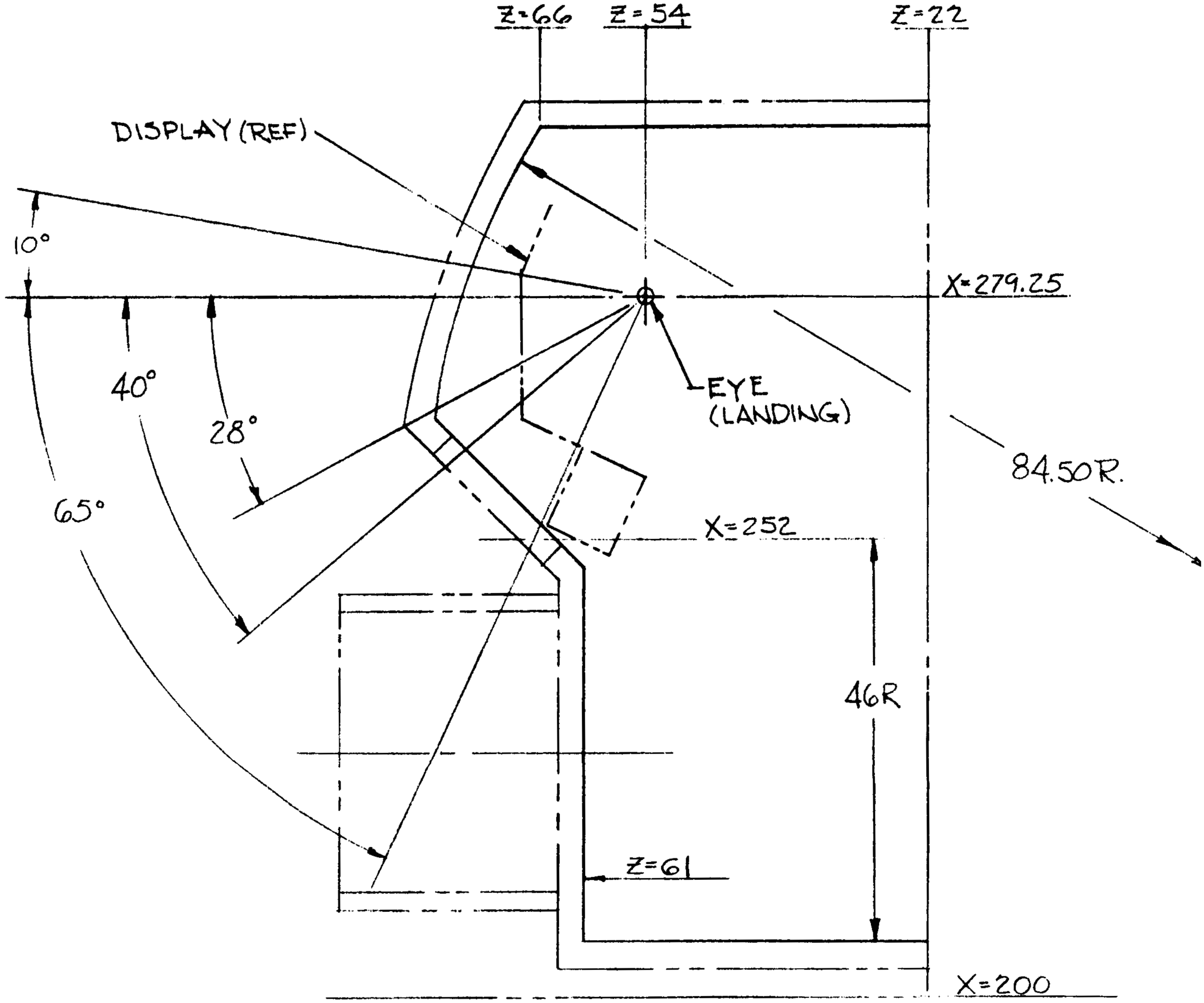


FIGURE 5.3 PRELIMINARY LEM CONTROL & DISPLAY PANELS FOR HOVER & LANDING PHASE

[redacted]

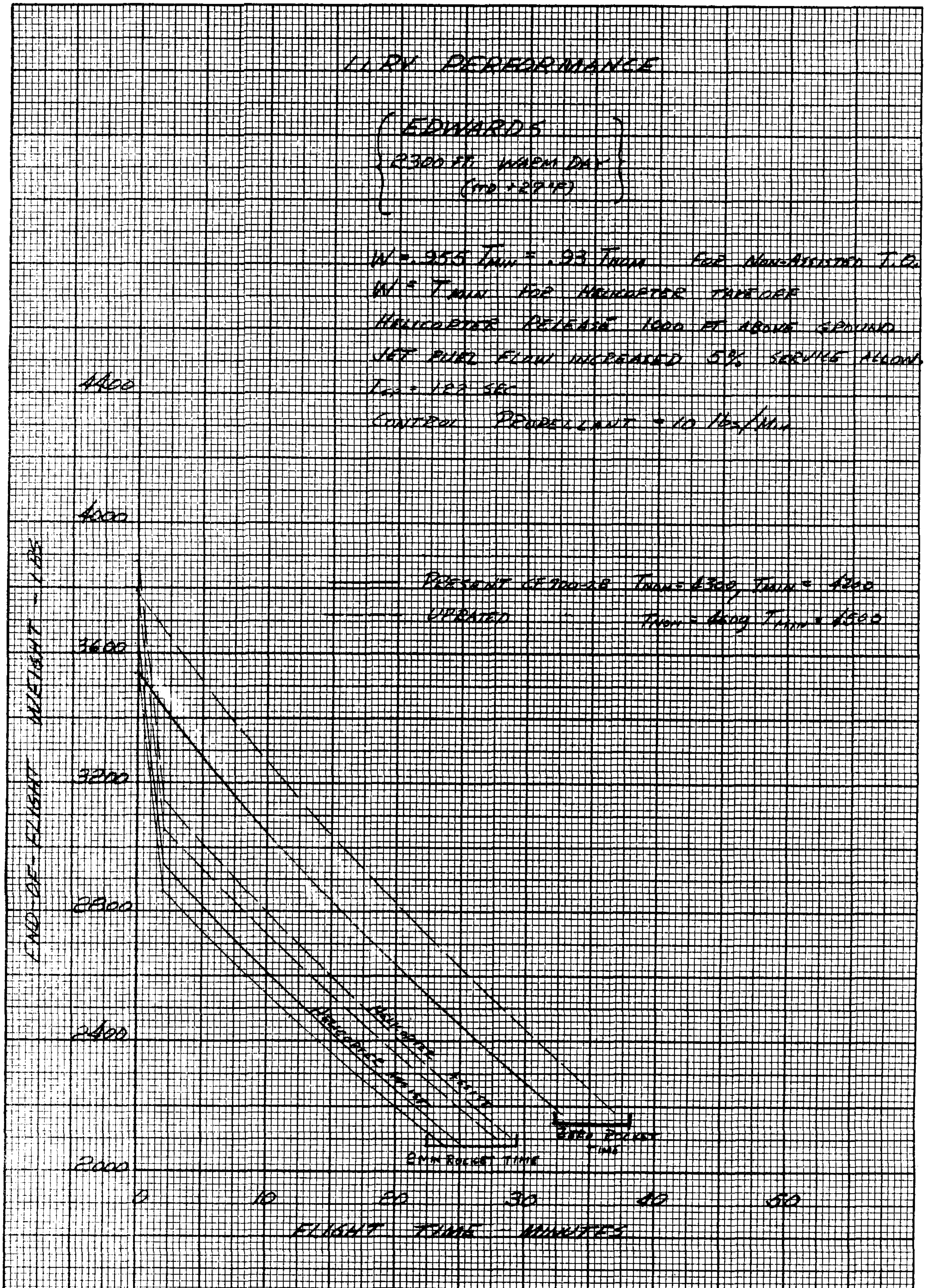
[redacted]



LEM Alternate Crew Capsule Geometry

Fig. 5.4

[redacted]



LLRV PERFORMANCE
EDWARDS
2300 FT WARM DAY
(STD +27°F)
W = .955 T MIN = .93 T MAX FOR NON-ASSISTED T.O.
W = T MIN FOR HELICOPTER TAKE OFF
HELICOPTER RELEASE 1000 FT ABOVE GROUND
JET FUEL FLOW INCREASED 5% SERVICE ALLOW.
CONTROL PROPELLANT = 10 lbs/MIN
PRESENT CF700-2B T MAX = 4300, T MIN = 4200
UPDATED T MAX = 4600, T MIN = 4500
HELICOPTER ASSIST
NON-ASSIST
2 MIN ROCKET TIME
ZERO ROCKET TIME
END-OF-FLIGHT WEIGHT - LBS
4400
4000
3600
3200
2800
2400
2000
0
10
20
30
40
50
FLIGHT TIME - MINUTES

Fig. 5.5

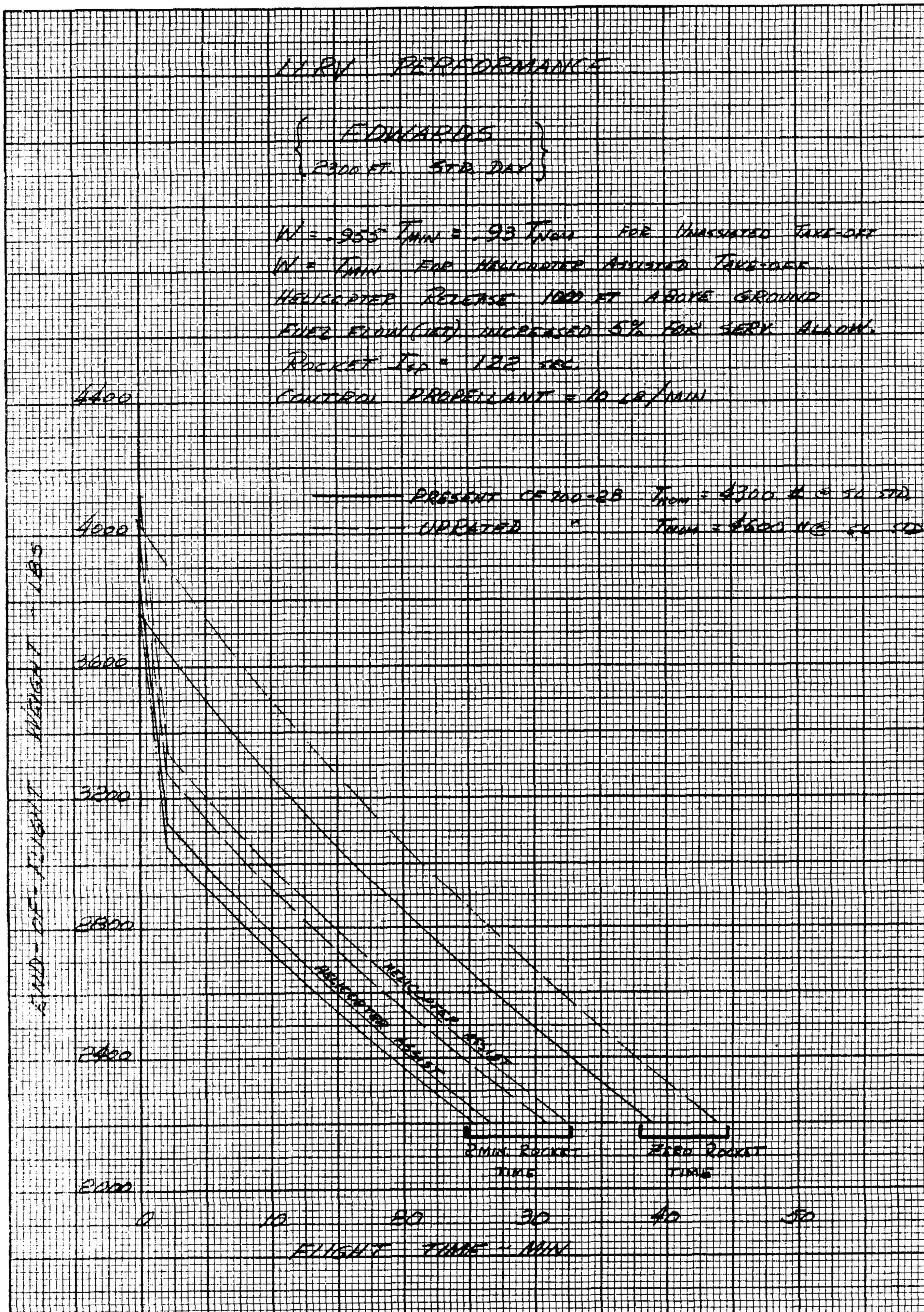

LLRV PERFORMANCE
EDWARDS
2300 FT. STD DAY
W = .955 T_MIN = .93 T_NOM FOR UNASSISTED TAKE-OFF
W = T_MIN FOR HELICOPTER ASSISTED TAKE-OFF
HELICOPTER RELEASE 1000 FT ABOVE GROUND
FUEL FLOW (JET) INCREASED 5% FOR SERV ALLOW.
ROCKET I_SP = 122 SEC.
CONTROL PROPELLANT = 10 LB/MIN
PRESENT CF 700-2B T_NOM = 4300 # @ SL STD
UPRATED " T_NOM = 4600 # @ SL STD
HELICOPTER ASSIST
HELICOPTER ASSIST
2 MIN. ROCKET TIME
ZERO ROCKET TIME
END-OF-FLIGHT WEIGHT ~ LBS
FLIGHT TIME ~ MIN
4400
4000
3600
3200
2800
2400
2000
0
10
20
30
40
50

Fig. 5.6

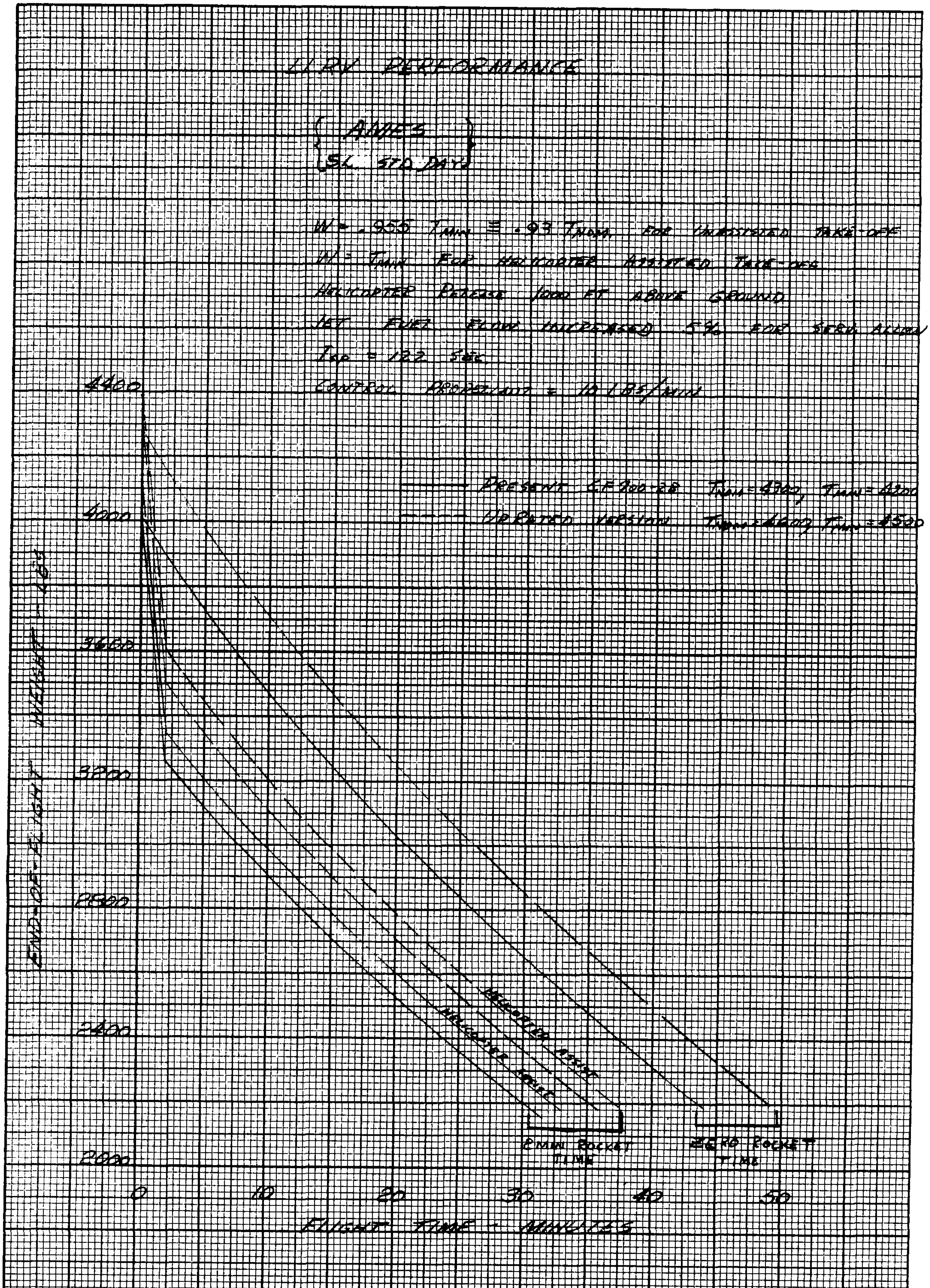

LLRV PERFORMANCE
AMES
SL STD DAY
W = .955 TMIN = .93 TNOM FOR UNASSISTED TAKE-OFF
W = TMIN FOR HELICOPTER ASSISTED TAKE-OFF
HELICOPTER RELEASE 1000 FT ABOVE GROUND
JET FUEL FLOW INCREASED 5% FOR SERV ALLOW
ISP = 122 SEC
CONTROL PROPELLANT = 10 LBS/MIN
PRESENT CF 700-2B TNOM = 4300, TMIN = 4200
UPRATED VERSION TNOM = 4600, TMIN = 4500
4400
4000
3600
3200
2800
2400
2000
HELICOPTER ASSIST
HELICOPTER ASSIST
8 MIN ROCKET TIME
ZERO ROCKET TIME
0
10
20
30
40
50
FLIGHT TIME - MINUTES
END-OF-FLIGHT WEIGHT - LBS

Fig. 5.7

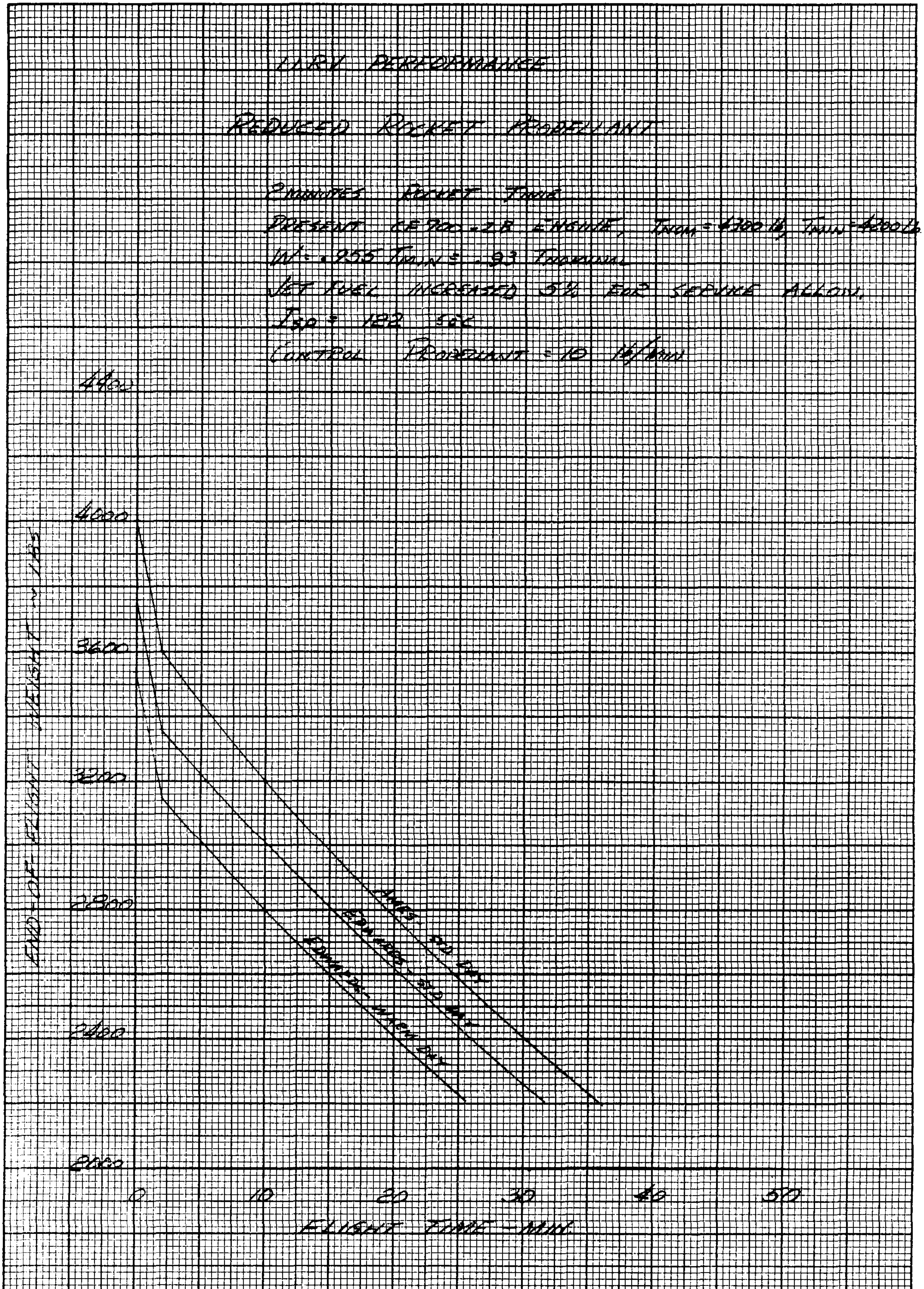

LLRV PERFORMANCE
REDUCED ROCKET PROPELLANT
2 MINUTES ROCKET TIME
JET FUEL INCREASED 5% FOR SERVICE ALLOW.
CONTROL PROPELLANT = 10 lb/MIN
AMES - STD DAY
EDWARDS - STD DAY
4400
4000
3600
3200
2800
2400
2000
0
10
20
30
40
50
FLIGHT TIME - MIN
END-OF-FLIGHT WEIGHT ~ LBS

Fig. 5.8

6. REFERENCES

1. "Grumman LTA-9 Feasibility Study Report" No. LED 470-1

2. MSC letter SLM-63-90, dated 23 April 1963 and enclosure "Minutes of Meeting, GAEC Presentation on LTA-9 Progress Summary"

3. "Feasibility Study for a Lunar Landing Flight Research Vehicle", Bell Aerosystems Company Report No. 7161-950001, March 8, 1962

4. "Lunar Landing Research Vehicle Progress Report No. 2", May 1, 1963

5. Phone Conversation between D. F. Gebhard (GAEC) and G. Metranga (LLRV Assistant Project Engineer - FRC), 9 May 1963

6. LLRV Work Statement

7. Memo LMO-500-20-013; Proposed Reaction Jet Command Logic for attitude control of LEM for maximum torque with combined control commands, 29 March 1963

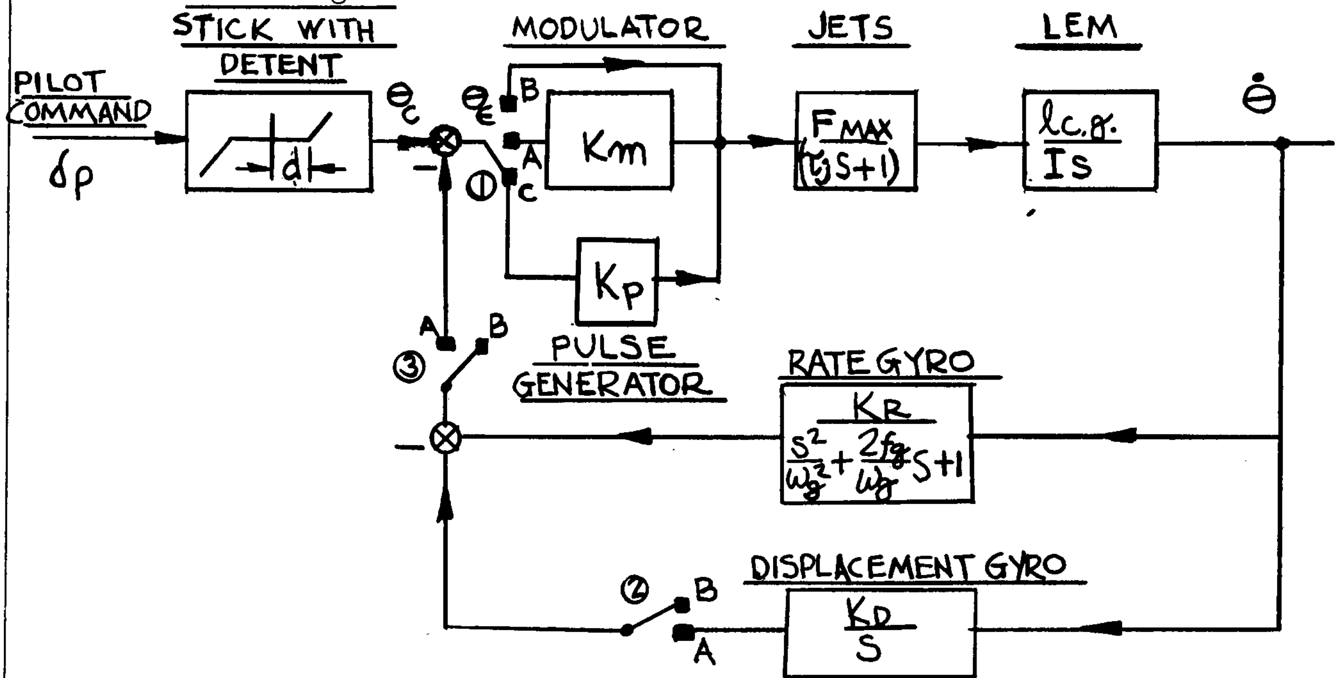
8. Project Development Plan, Free-Flight Lunar Landing, and Take-Off Research Vehicle, Office of Manned Space Flight, NASA Headquarters, Flight Research Center, Edwards, California, December 1962 (Second Revision).

9. GAEC Report No. LPL-480-1, Training Plan for the Lunar Excursion Module, Vol. I - Flight Crew Training, dated 15 May 1963.

APPENDIX A

LEM ATTITUDE CONTROL CONFIGURATION

A. Block Diagram

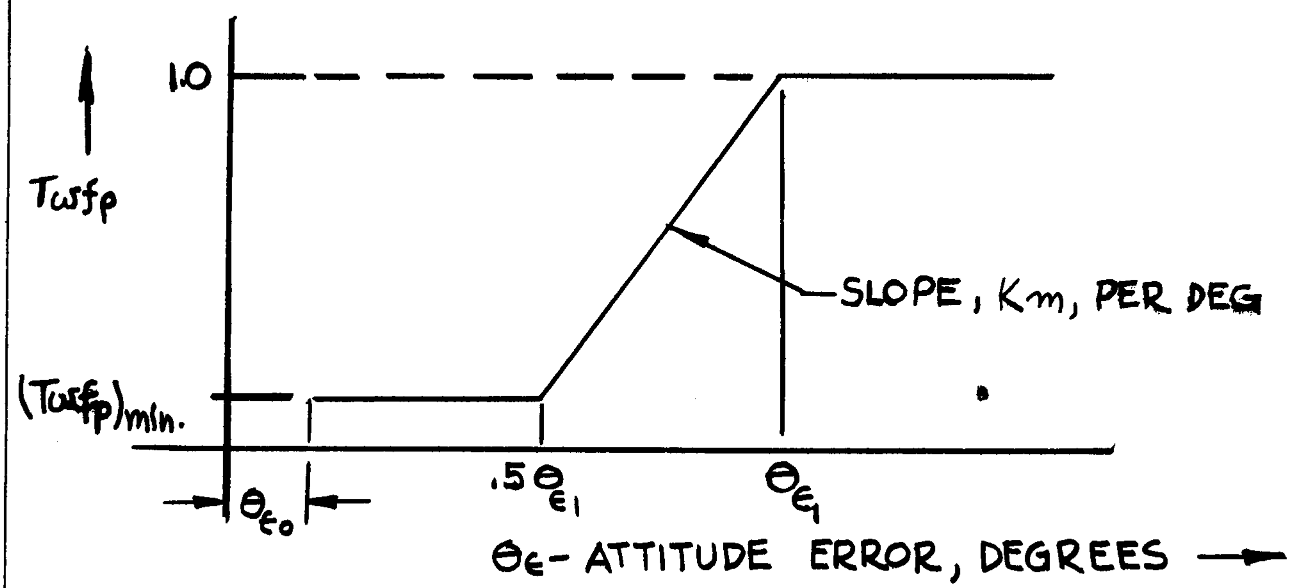


B. LEM RCS Operating Modes

LEM RCS Operating Modes	Switch ①	②	③
1. Rate-Command			
a. $\delta_p > d$, W/O Attitude - hold	A	B	A
b. $\delta_p > d$, with Attitude - hold	A	A	A
2. Attitude Command	A	A	A
3. Emergency			
a. Manual - Direct, on-off	B	B	B
b. Manual - Direct, minimum impulse bit, (.6 lb-sec. at 2-5 p.p.s.)	C	B	B

C. Modulator Characteristics:

Piecewise - Linear, Pulse - Width Modulation:



D. Mode Response Characteristics

1. Rate Command - the LEM RCS in this mode will respond to pilot input commands as a first order system. The time response, defined as a function of system gains, is:

$$\tau = I/K_m K_R F l_{c.g.}$$

2. Attitude - Hold and Command - the LEM RCS in this mode will respond to pilot input commands as a second-order system. The natural frequency and damping ratio, as a function of system gains, is:

$$\omega = \frac{K_m K_D F l_{c.g.}}{I}$$

$$\zeta = \frac{1}{2\omega} \frac{K_m K_R F l_{c.g.}}{I}$$

[redacted]

E. Required LLRV Capability on LEM RCS Characteristics

K_R : Rate feedback gain .1 to 1.5 V/r.p.s.
0.5 V/r.p.s. nominal

K_D : Attitude feedback gain .25 to 1.0 V/rad.
1.0 V/rad nominal

K_m : Modulator gain
.05 to 5 per deg. attitude error, .55 per deg. attitude error nominal

θ_{ϵ_0} : Modulator dead-zone
$\pm$ .05 to 0.5 deg. of attitude error

θ_{ϵ_1} : Attitude error when Twfp = 1 G $\pm$.5 to 2 deg.

f_p : Modulator pulse repetition frequency, 5 - 10 pulses per sec.

T_ω : Modulator pulse width, .006 sec. - .012 sec., for F max. = 100 lbs. - .6 lbs. - sec. to 1.2 lb. - sec.

τ : Rate-command response time constant, .1 to 1.5 sec., 0.4 sec. nominal

ω, ζ : Attitude-command response natural frequency and damping,
0.5 - 20. r.p.s., 2.50 r.p.s. nominal
.3 - .7 ratio, .63 nominal

$\omega_g\ \zeta_g$: Rate Gyro response natural frequency and damping
100 r.p.s. and 0.7 ratio nominal

$Flc.g./I$: Control power, r.p.s., as specified in Section 3.1.1

τ_J : Jet thrust build-up, 7 - 10 m sec. nominal

[redacted]

APPENDIX B

LEM TERMINAL DESCENT AND LANDING PERFORMANCE ENVELOPE

At the present time the flight paths and maneuvers to be used during the terminal descent and landing are still under investigation. Hence, rather than arbitrarily selecting discrete flight paths and maneuvers to form the basis for LEM Flight Simulator requirements, a nominal descent trajectory was presented in Figure 3-2. This analysis is presented as a further aid in establishing simulator requirements. It is based on an idealized set of assumptions regarding the piloting skill and includes a series of performance calculations indicating the maximum performance which might be obtained.

Discussion of Analysis

Assumptions:

1. For the flight period under consideration, the fuel mass change is low enough so that the LEM mass changes can be neglected.

2. The vertical (perpendicular to the horizon) acceleration will be constant. This assumption simplifies the analysis and facilities constant line-of-sight operation as will be seen later. It should be noted that the horizontal translations at constant attitude which are common to most VTOL operation also require constant vertical acceleration.

3. The LEM tilt angle can be changed instantaneously.*

4. Orbital mechanics can be neglected.

5. At touchdown both the horizontal and descent velocity are zero.

The following two cases will be treated:

Case I. A vertical descent from a given altitude, with an initial descent rate to be chosen as a result of the analysis.

Case II. An equal period of horizontal acceleration and deceleration starting at a given altitude with zero horizontal velocity and a vertical descent rate to be chosen as a result of the analysis.

* Maneuver time will be very small compared to flight times.

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Symbols And Typical LEM Parameters:

F - Fuel required for entire flight, slugs.

g - Lunar gravitational acceleration, ft/sec.^{2} (5.4).

G - Earth gravitational acceleration, ft/sec.^{2} (32.2).

h - LEM altitude from lunar surface, positive up, ft.

I - LEM descent engine specific impulse, sec. (290).

l - LEM horizontal distance from starting point, positive in direction of orbital flight, ft.

m - Fuel mass flow, slugs/sec.

M - LEM mass, slugs. (310).

R - Total range from starting point, ft.

t - Time required for descent, sec.

T - Total descent engine thrust, positive up, earth lbs.

α - Descent engine thrust axis tilt from the vertical on the earth, positive nose up, rad.

ϵ - Final flight path slope, positive down, rad.

θ - Descent engine thrust axis tilt from the vertical on the moon, positive nose up, rad.

ϕ - Sight angle measured from LEM Z axis, positive down, rad.

o - (as subscript) Initial Value.

Dot over symbol indicates differential with respect to time.

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Case I.

Summing forces perpendicular to the horizon:

$$M\,(g + \ddot{h}) = I\,G\,\dot{m} \qquad (1)$$

For a constant vertical acceleration

$$\dot{h}_o^{\,2} = 2\,\ddot{h}\,h_o \qquad (2)$$

and

$$t = \frac{2\,h_o}{\dot{h}_o} \qquad (3)$$

The fuel used for descent

$$F = \dot{m}\,t \qquad (4)$$

Combining equations (1), (2), (3) and (4)

$$F = \frac{2\,h_o\,M\,g}{I\,G\,\dot{h}_o} + \frac{\dot{h}_o\,M}{I\,G} \qquad (5)$$

This equation is plotted in Figure B-1 for the typical LEM parameters given on Page 3-2 , and an initial altitude of 1,000 feet.

The figure shows a distinct minimum fuel value of 220 pounds occurring at an initial descent rate of 104 ft/sec. Physically this means that for lower descent rates it takes too much time to get down and for higher descent rates there is too much kinetic energy to be cancelled.

The expression for the minimum fuel is obtained by differentiating equation (5) and equating the derivative to zero.

$$2\,h_o\,g = \dot{h}_o^2 \qquad (6)$$

Note that if $\dot{h}_o$ is considered as the final rather than the initial velocity, this is the equation for a mass falling a distance h_o under a gravitational acceleration g. Hence the optimum descent rate for vertical descent from any altitude is the velocity of the LEM at that altitude if gravity were reversed and it had falled away from the moon.

Using equation (6) in (5), the fuel/LEM mass ratio is (7)

$$\frac{F}{M} = \frac{\sqrt{8\,h_o\,g}}{I\,G} \qquad (7)$$

Case II.

For a constant vertical deceleration

$$2\left(\frac{T\cos\Theta}{M} - g\right)h_o = \dot{h}_o^2 \qquad (8)$$

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For a constant horizontal acceleration and deceleration of equal period

$$R = \frac{T}{M} \sin \Theta \, \frac{t^2}{4} \tag{9}$$

The rocket impulse is:

$$t \, T = I \, F \, G \tag{10}$$

Combining equations (3), (9) and (10)

$$\sin \Theta = \frac{2 \, M \, R \, \dot{h}_o}{I \, F \, h_o \, G} \tag{11}$$

and combining equations (3) and (8)

$$\cos \Theta = \frac{\dot{h}_o \, M}{I \, F \, G} + \frac{2 \, M \, h_o \, g}{I \, F \, \dot{h}_o \, G} \tag{12}$$

Squaring and adding equations (11) and (12)

$$\frac{4M^2 \, R^2 \, \dot{h}_o^2}{G^2 \, I^2 \, F^2 \, h_o^2} + \frac{\dot{h}_o^2 \, M^2}{I^2 \, F^2 \, G^2} + \frac{4 \, M^2 \, h_o \, g}{I^2 \, F^2 \, G^2} + \frac{4 \, M^2 \, h_o^2 \, g^2}{I^2 \, F^2 \, \dot{h}_o^2 \, G^2} = 1 \tag{13}$$

or $$R^2 = \frac{I^2 \, F^2 \, G^2 \, h_o^2}{4 \, M^2 \, \dot{h}_o^2} - \frac{h_o^2}{4} - \frac{h_o^3 \, g}{\dot{h}_o^2} - \frac{h_o^4 \, g^2}{\dot{h}_o^4}$$

This equation is plotted in Figure B-2 for 714 lbs. of fuel ($\Delta V = 650$ ft/sec.)

Differentiating with respect to $\dot{h}_o$ and equating the result to zero, the optimum descent rate

$$\dot{h}_o = \left[\frac{4 \, h_o^2 \, g^2}{\dfrac{I^2 \, F^2 \, G^2}{2 \, M^2} - 2 \, h_o \, g} \right]^{\frac{1}{2}}$$

Since

$$\tan \epsilon = \frac{T \cos \Theta - M \, g}{T \sin \Theta} \tag{15}$$

We can write, using equations (3), (10), (11), and (12)

$$\tan \epsilon = \frac{h_o}{2 \, R} \tag{16}$$

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and

$$\phi = \theta + \epsilon \qquad (17)$$

The maximum horizontal velocity is obtained as follows:

$$\dot{l}_{max} = \frac{T \sin \theta}{M} \frac{t}{2} \qquad (18)$$

and, using equations (3), (10) and (11)

$$\dot{l}_{max} = \frac{R \ddot{h}_o}{h_o} \qquad (19)$$

The above equations form the basis for figures B-2 to B-7. First equation (14) was used to obtain the variation of the optimum initial descent rate with fuel weight for the typical LEM parameters given on Page B-2. . (See Figure B-3). Then these combinations of fuel and initial descent rate were used in equation (13) to obtain the range variation with hover time at the landing site for a given fuel weight or Δ V. (See Figure B-4). Finally the ranges and fuels from this calculation were used in equations (9),(10), (11), (16), (17), and (18) to obtain the variation of tilt angle, final glide slope, final sight angle, maximum velocity, and flight duration variation with range. (Figures B-5 and B-6).

Since tilt angles as high as 43 degrees are indicated, the following analysis was performed to gain some insight regarding the difficulty of such maneuvers on the moon.

For a constant throttle setting the vertical acceleration resulting from vehicle tilt on the moon is:

$$\ddot{h} = \frac{G}{6} (1- \cos \theta) \qquad (20)$$

and the vertical acceleration on the earth is:

$$\ddot{h} = G (1-\cos \alpha) \qquad (21)$$

For the same acceleration to occur in each case the tilt angles are related by:

$$5 + \cos \alpha = 6 \cos \theta \qquad (22)$$

This equation is plotted in Figure B-7.

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Discussion of Calculation Results

The variation of range with initial descent rate for a fixed fuel load is shown in Figure B-2. It indicates that range capability is seriously compromised by descent rates below the optimum value, whereas for descent rates above the optimum the range reduction is much smaller. Reference to Figure B-3, which shows the optimum initial descent rate variation with range (or fuel), indicates that for the longer ranges the optimum initial descent rate is about 30 ft/sec. and varies little with range. Hence it can be concluded that the LEM should be descending at a rate somewhat higher than this value when it is at 1,000 feet attitude.

The trade-off of range with hover time given in Figure B-4 shows that ranges much above the 1,000 feet value can be achieved, while still providing extensive low speed maneuvering time at the landing site. One rational for selection of a design range based on this plot would be to choose a combination of range and hover time such that the product of the two is a maximum. Such an approach would result in a range of 3,500 feet with a hover time of 43 seconds. This range would permit a larger foot print than the nominal 1,000 feet value and the 43 second hover time would provide an adequate margin for final maneuver adjustments.

The variation of optimum tilt angle with range shown in Figure B-5 indicates that even for large ranges the tilt angle does not exceed 45 degrees. Hence a LEM simulator like the LLRV could duplicate the LEM performance if it had this tilt capacity after compensating for extraneous forces.

It also should be noted that for all but the lowest ranges the final line of sight to the landing site is nearly constant. As a result the pilot could use a fixed window reference point for the final result regardless of the range he wants to achieve.

Figure B-7 shows the angle a VTOL on earth would have to reach so that, with the throttle held constant, it accelerates downward at the same rate that the LEM would achieve for a given tilt angle on the moon. For the 45 degree tilt noted above as the maximum value needed to realize the maximum LEM performance, the corresponding earth angle is 18 degrees. This angle is below the tilt angles commonly used by earth VTOL pilots on earth. Thus the lunar tilt angles suggested by this analysis should present no difficult throttle coordination problems.

The variation of maximum horizontal velocity with range is shown in Figure B-6. For the design range discussed previously the maximum velocity is 100 ft/sec, which is well within helicopter speeds.

The variation of translation flight duration with range is shown in B-8, and suggests that assumption 3 is valid for the longer ranges since the time required to tilt the LEM 45 degrees the required four times will be small compared to the flight time.

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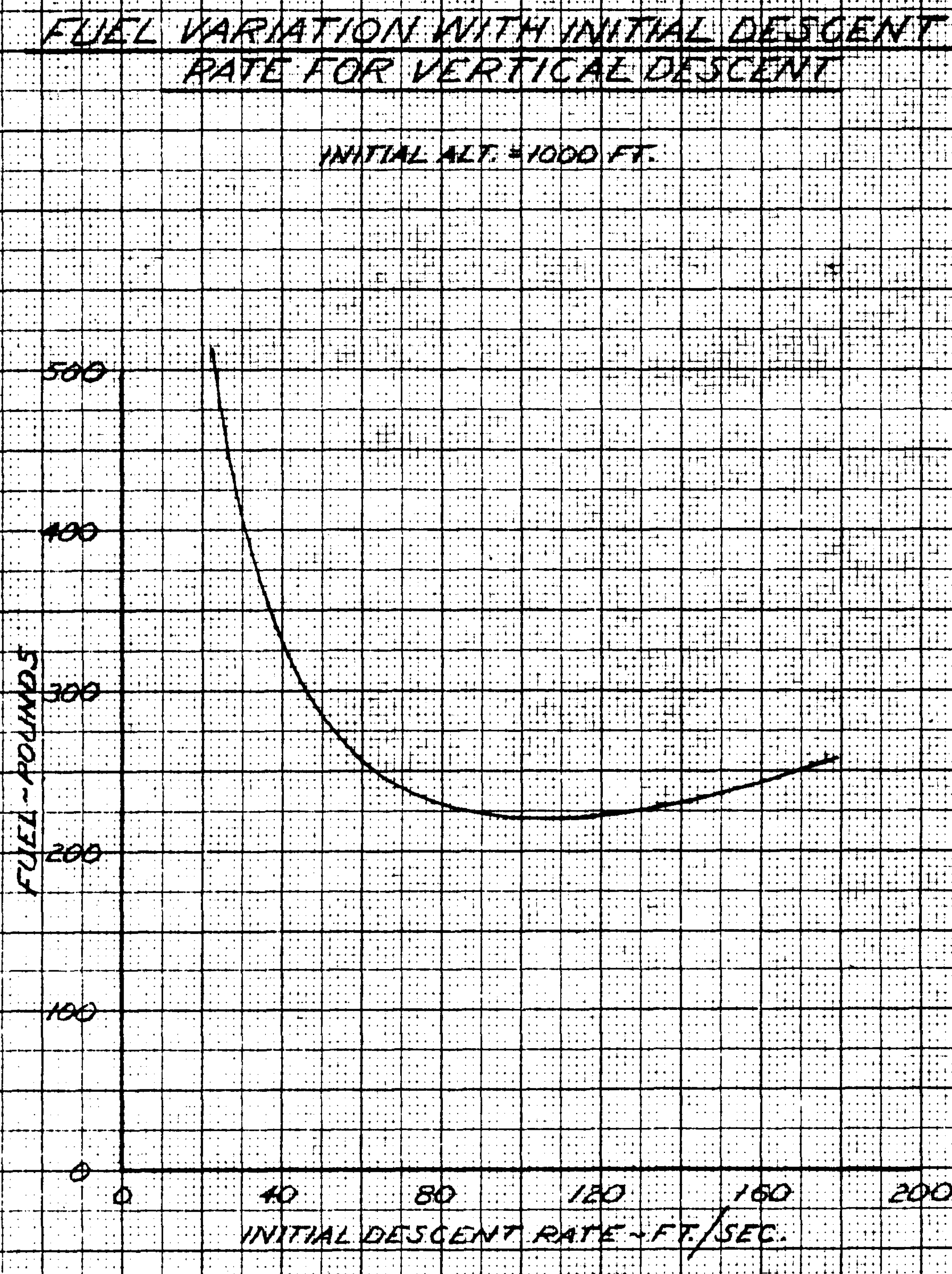
FUEL VARIATION WITH INITIAL DESCENT RATE FOR VERTICAL DESCENT
INITIAL ALT. = 1000 FT.
FUEL ~ POUNDS
500
400
300
200
100
0
0
40
80
120
160
200
INITIAL DESCENT RATE ~ FT./SEC.

Fig. B-1

RANGE VARIATION WITH INITIAL DESCENT RATE FOR CONSTANT FUEL LOAD

ΔV = 650 FT. SEC.
INITIAL ALT. = 1000 FT.

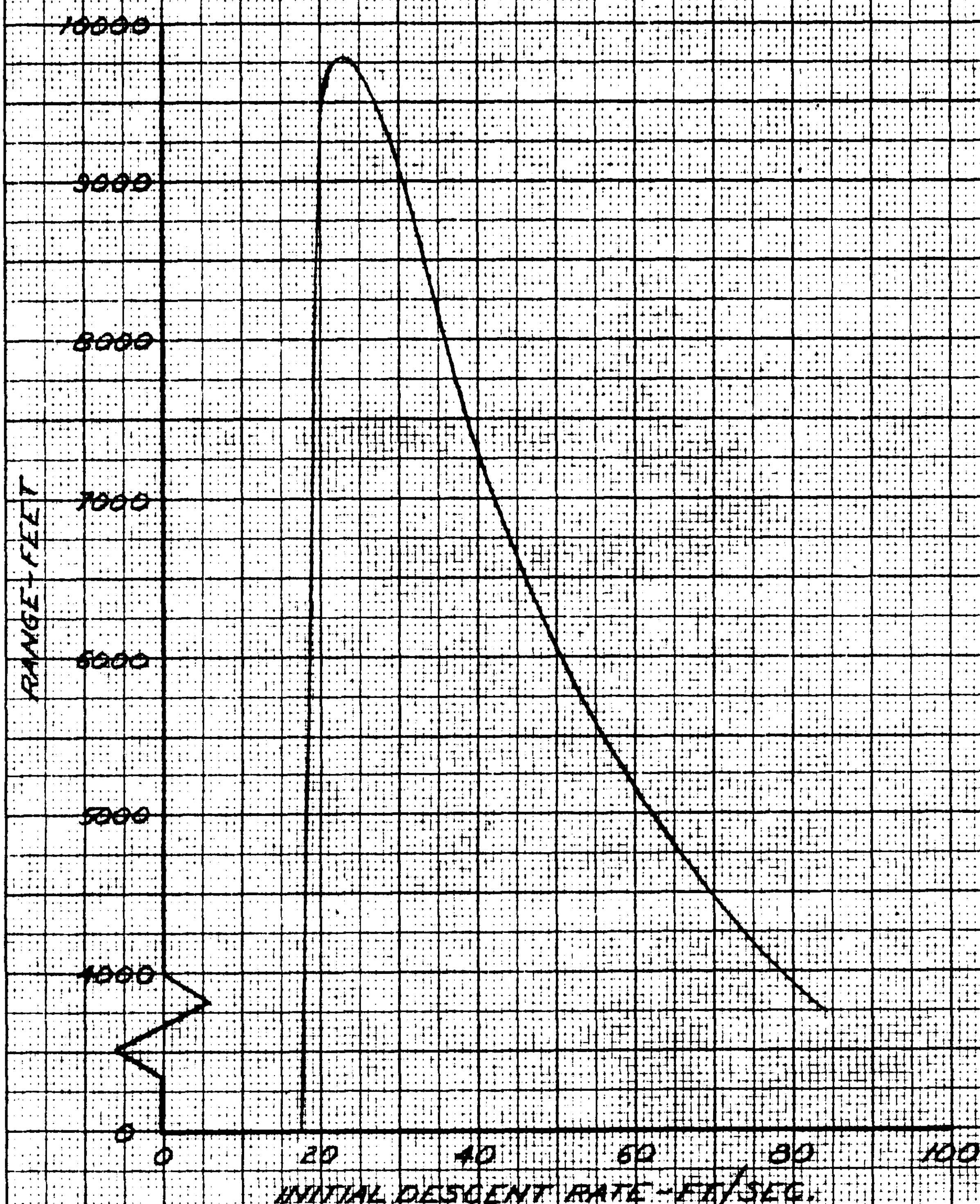

Fig. B-2

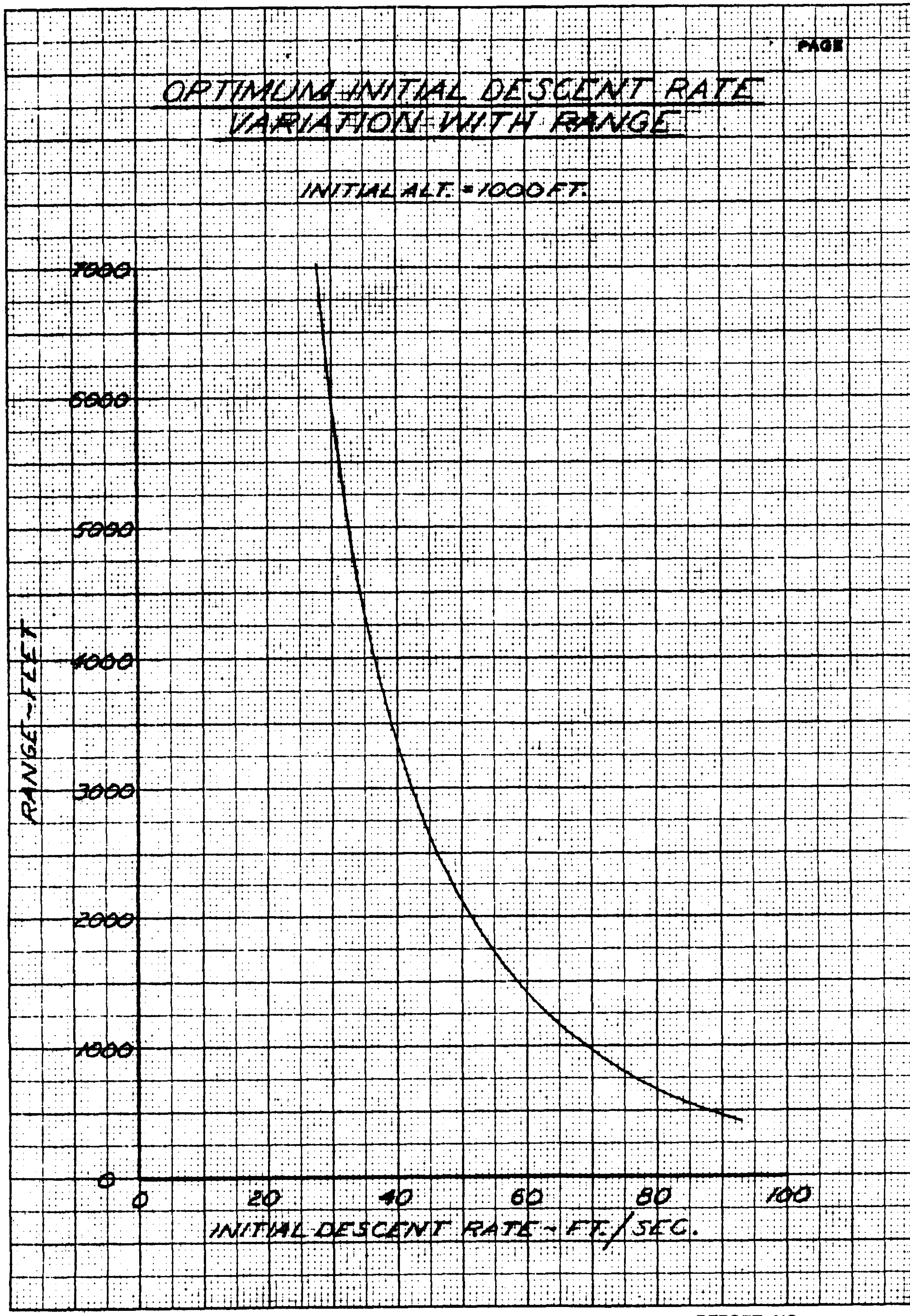

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Fig. B-3

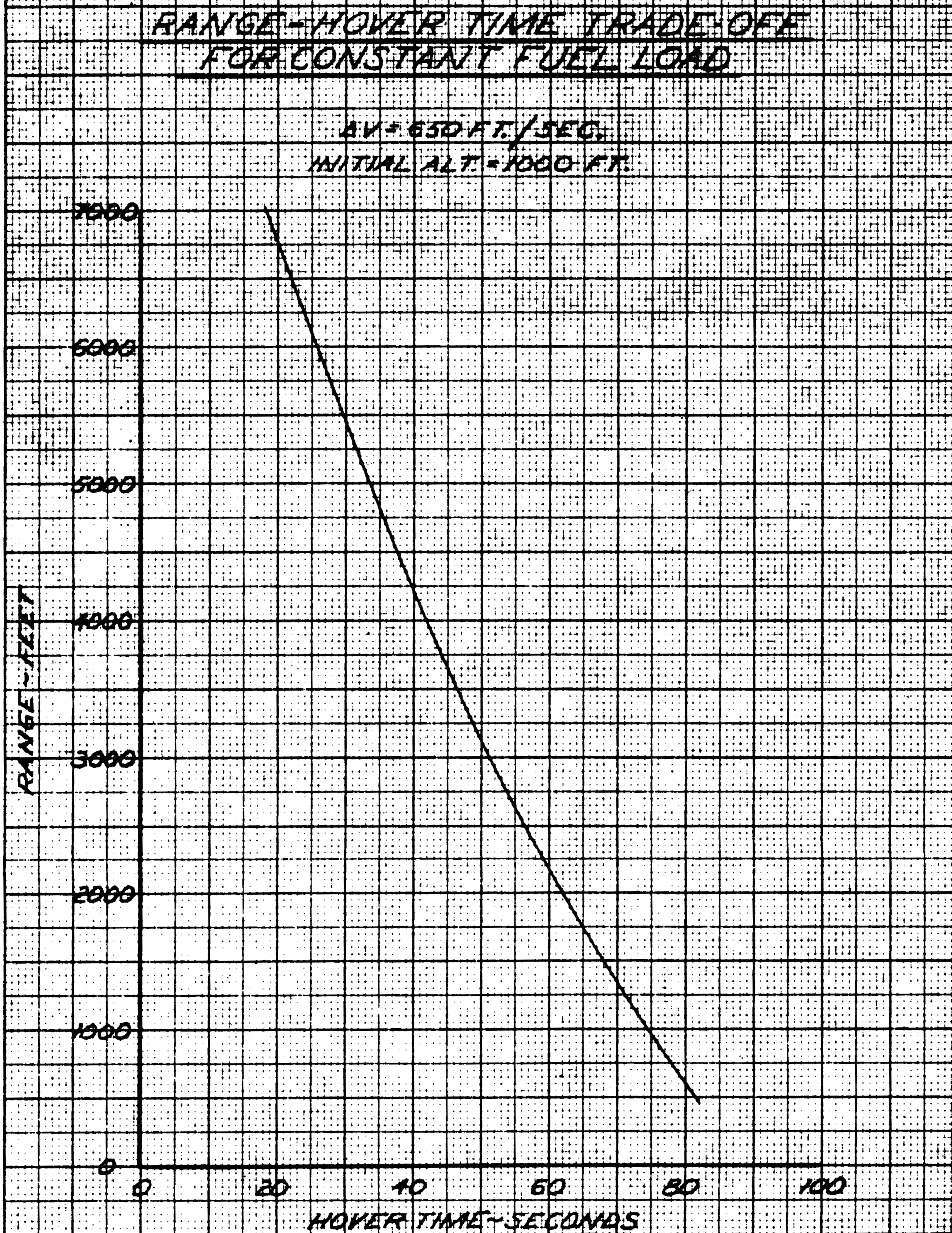

RANGE-HOVER TIME TRADE-OFF
FOR CONSTANT FUEL LOAD
ΔV = 650 FT./SEC.
INITIAL ALT. = 1000 FT.
7000
6000
5000
4000
3000
2000
1000
0
RANGE ~ FEET
0
20
40
60
80
100
HOVER TIME ~ SECONDS

Fig. B-4

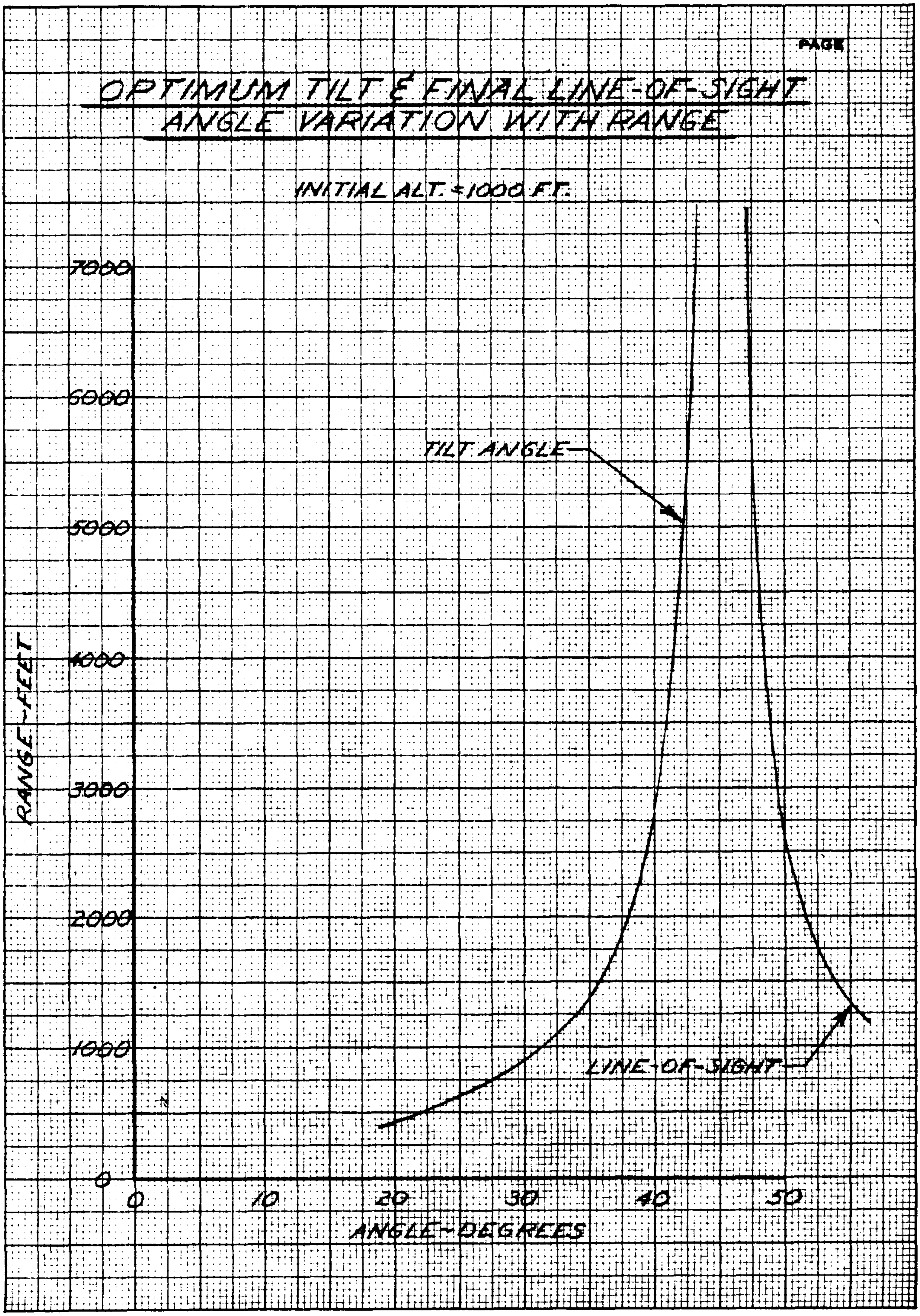

OPTIMUM TILT & FINAL LINE-OF-SIGHT ANGLE VARIATION WITH RANGE
INITIAL ALT. = 1000 FT.
TILT ANGLE
LINE-OF-SIGHT
RANGE - FEET
0
1000
2000
3000
4000
5000
6000
7000
0
10
20
30
40
50
ANGLE - DEGREES
PAGE

Fig. B-5

MAXIMUM HORIZONTAL VELOCITY VARIATION WITH RANGE

INITIAL ALT. = 1000 FT.

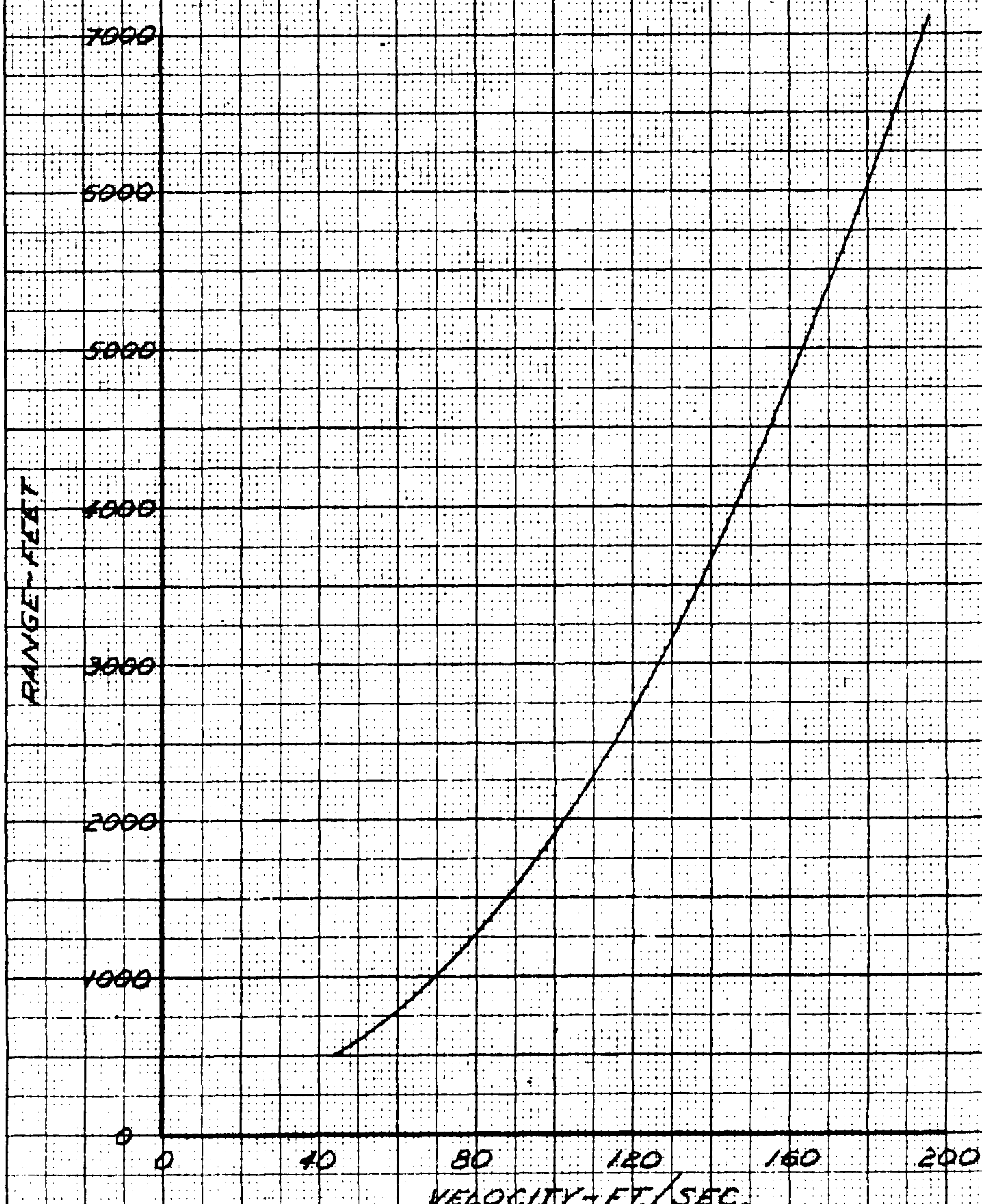


Fig. D-6

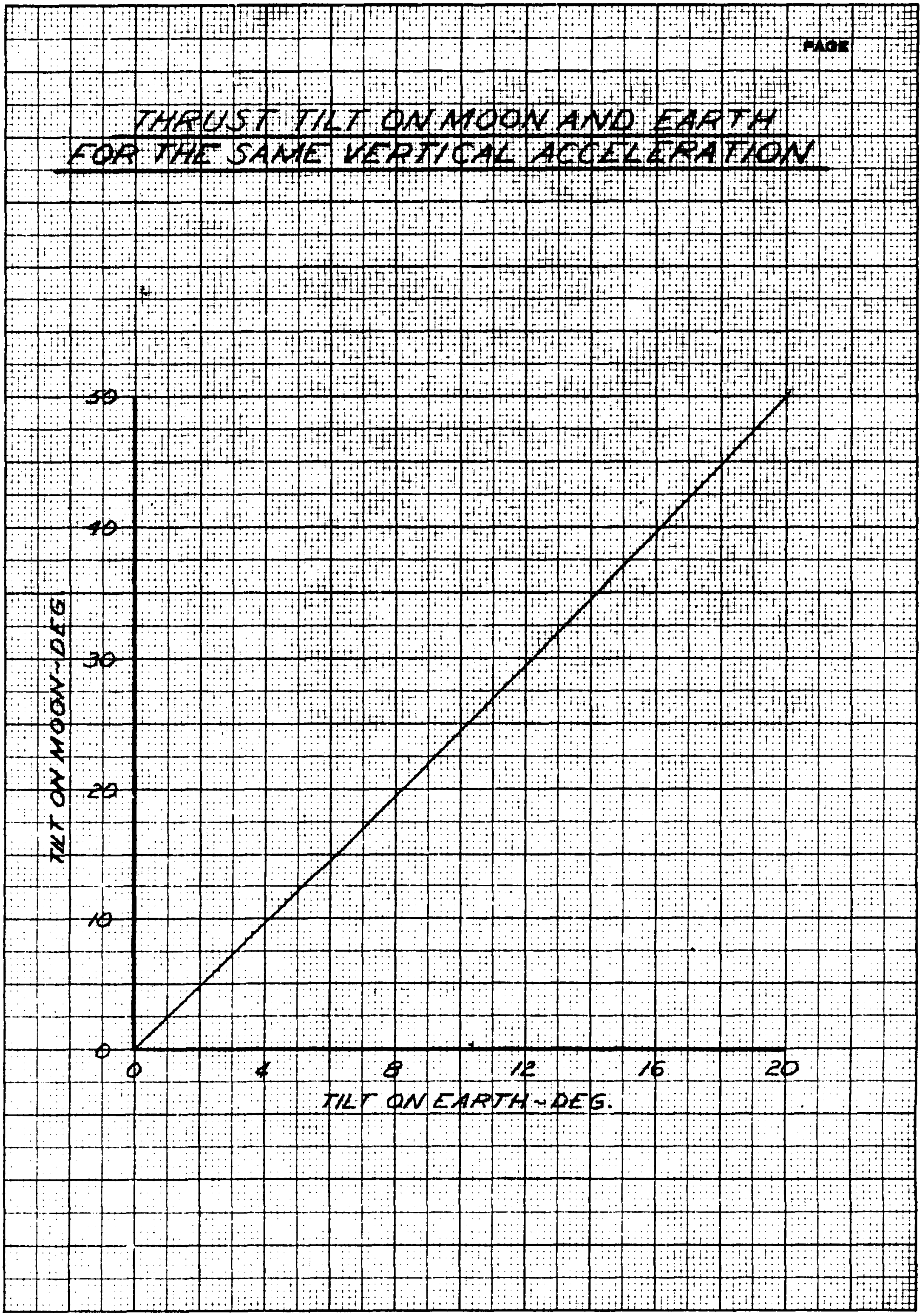

THRUST TILT ON MOON AND EARTH
FOR THE SAME VERTICAL ACCELERATION
TILT ON MOON ~ DEG.
50
40
30
20
10
0
0
4
8
12
16
20
TILT ON EARTH ~ DEG.

Fig. B-7

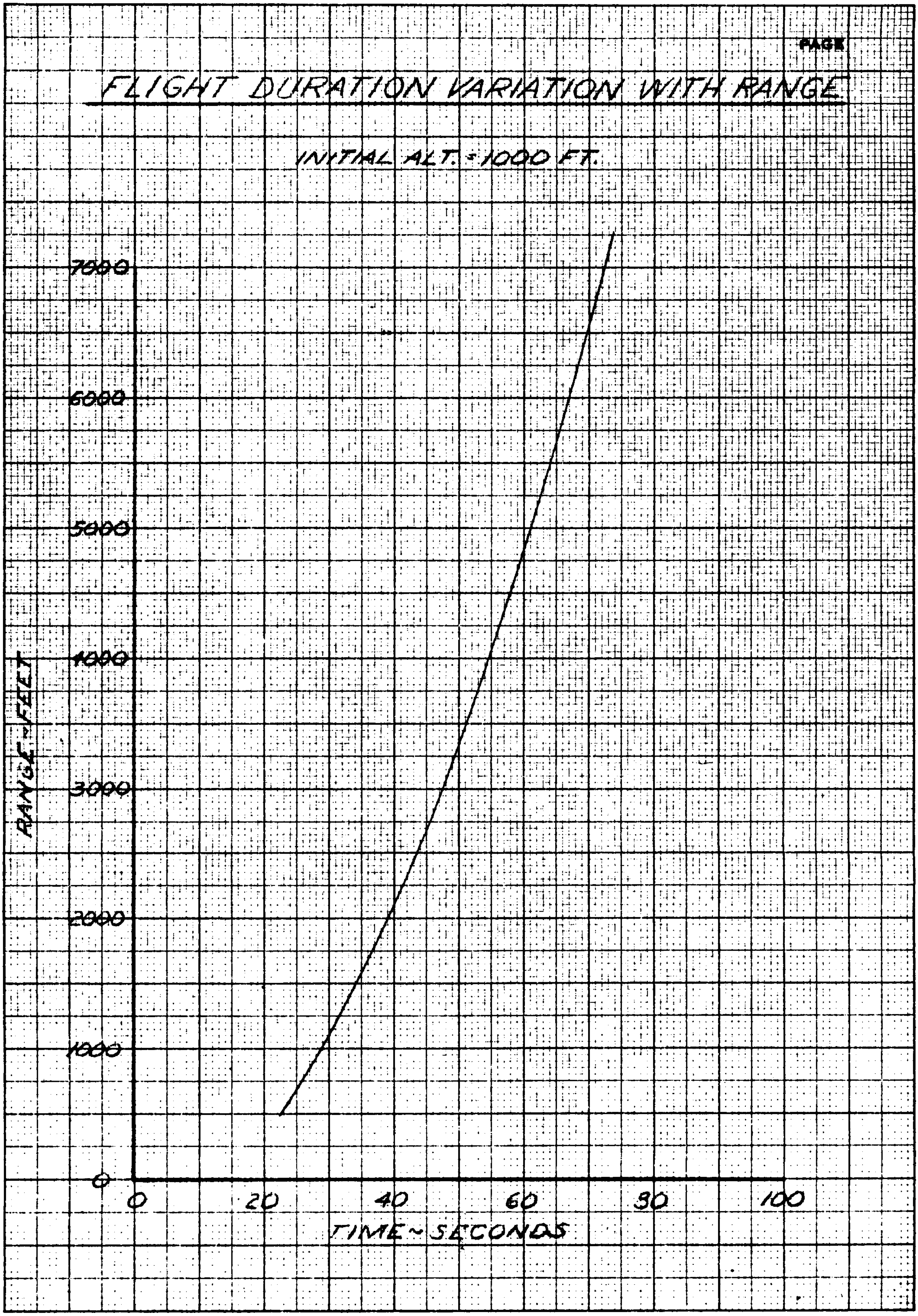

PAGE
FLIGHT DURATION VARIATION WITH RANGE
INITIAL ALT. = 1000 FT.
RANGE ~ FEET
7000
6000
5000
4000
3000
2000
1000
0
0
20
40
60
80
100
TIME ~ SECONDS

Fig. B-8

APPENDIX C

SUGGESTED MODIFICATION TO THE AUTOMATIC ENGINE CONTROL SYSTEM

As now designed, the LLRV jet engine is required to apply a force vector at the vehicle center of gravity which cancels the gravitational force difference between the earth and the moon (approximately 5/6 of the vehicle weight) and any aerodynamic forces due to vehicle motion. To achieve this the engine is mounted in a gimbal assembly at the center of gravity and is controlled by an automatic system which computes the proper throttle setting and tilt angles and drives the engine power lever and gimbal assembly to these values. A pair of hydrogen peroxide rocket engines are installed symmetrically about the vehicle vertical body axis to provide the same forces on the vehicle in earth flight as on the moon. The specific impulse of the hydrogen peroxide is only 40 per cent of the LEM propellants, however, and so a proportionally greater amount of fuel must be carried on the LLRV to achieve LEM performance.

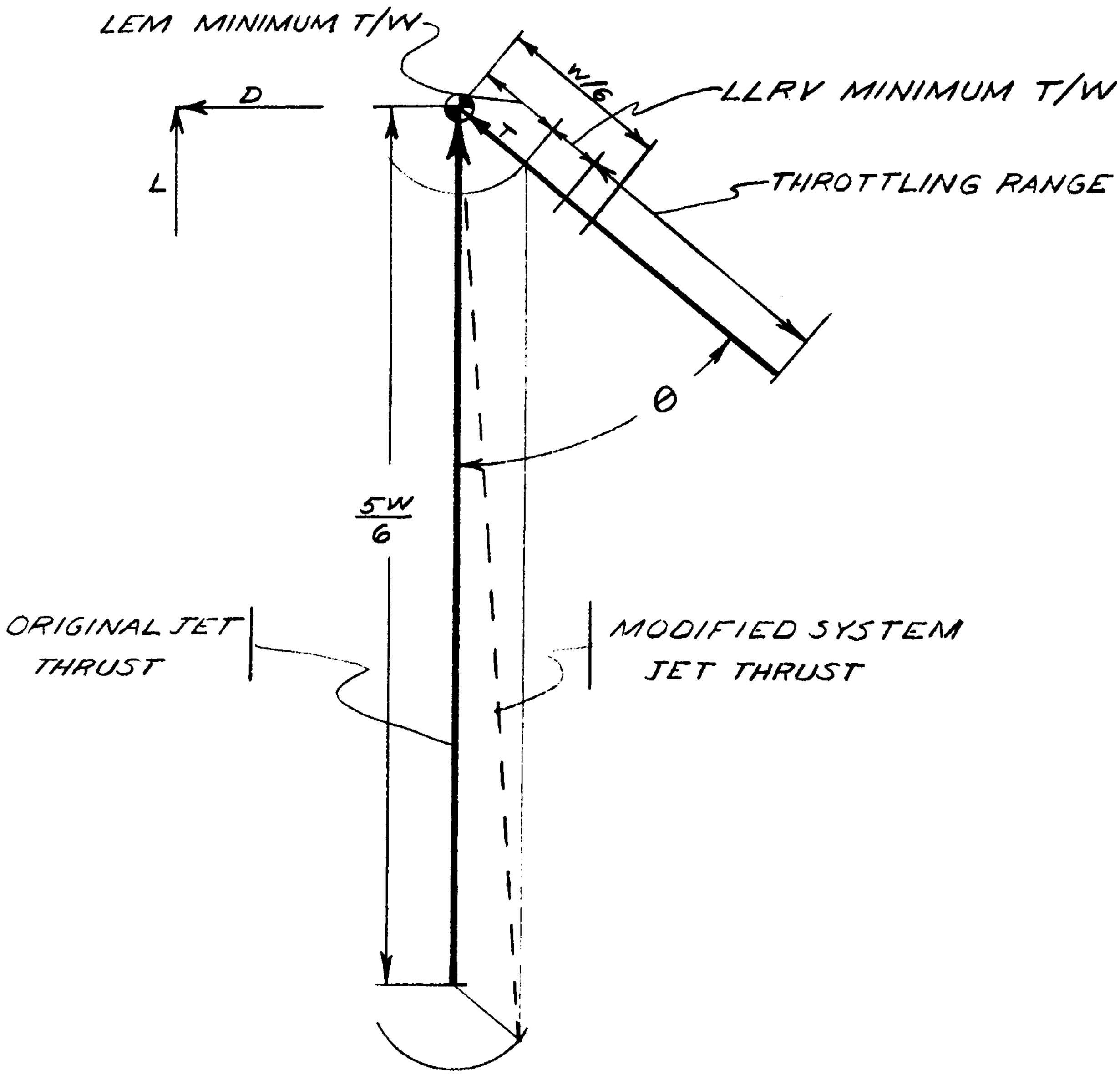
In order to reduce the hydrogen peroxide required and yet retain rocket type handling qualities, it is suggested that only the portion of the total rocket thrust over which the pilot exercises control be provided by the actual rocket, while the remainder is provided by the jet engine. This scheme is illustrated vectorially in Figure C-1, for typical LEM parameters. It is assumed for this figure that the system must accommodate LEM tilt angles up to 50 degrees and that the fully throttled LEM descent engine will provide a thrust/weight ratio of .5 on the moon.

If T is the minimum rocket thrust and θ is the LEM tilt angle, then the computed longitudinal aerodynamic force to be canceled (D) should be decreased by T sin θ; and the computed vertical aerodynamic force to be canceled (L) should be decreased by T cos θ. Since T is a preset constant and θ is already being computed by the system as now designed, generating the addition to the aerodynamic force error should be a straight forward resolving function.

Since the LLRV lift rocket engine also has a minimum thrust level, use of the above system limits the pilot to a minimum lunar thrust/weight ratio which is the sum of that for the LEM and the LLRV. The LLRV minimum value is .333 for a gross weight of 3600 earth pounds. Hence the operational limit is .833, based on a minimum LEM value of .5. Using these numbers the hydrogen peroxide requirements could be reduced by half while still retaining a minimum thrust/weight ratio well below 1.0. If it was desired to achieve the actual LEM minimum thrust/weight ratio of .5, the hydrogen peroxide could be reduced by 16 per cent. Aside from the rocket propellant savings (up to 300 pounds) obtained with a minimum of added complication, the approach offers the following advantages:

Larger available tilt angle - the jet engine now follows the vehicle tilt to a greater extent.

Closer approximation to mass changes due to fuel burn-off.



Vector Diagram of LLRV Automatic Control System Modification

Fig. C-1

APPENDIX D

PRELIMINARY INVESTIGATION OF EMERGENCY DESCENT RECOVERY REQUIREMENTS

Since crew and vehicle safety are of paramount importance for the LLRV application, adequate provisions must be made for a safe descent of the vehicle and its occupants in case of an emergency or malfunction. Two separate modes of recovery were studied. In one, the crew departs from the vehicle via ejection seats, after which a drag chute deploys to sustain the vehicle. Just prior to touchdown the vehicle is decelerated to a safe landing sink rate by a short duration JATO bottle that is automatically energized at the proper altitude.

In the second approach, the crew rides down with the vehicle. Again the drag chute is deployed if the vehicle is above the minimum required altitude (200 to 250 feet for a Stencel-type parachute) and the vehicle emergency rocket system is actuated at the appropriate altitude to decelerate the vehicle for landing. Below this altitude, peroxide rockets replace the parachute.

A study was made to determine the time, distance, and fuel required to decelerate the LLRV to a safe landing velocity. In the analysis that follows it is assumed that the drag chute has been deployed and the vehicle is sinking at some velocity before the rockets are fired. The numerical solutions selected are for the special case where the vehicle is sinking at the terminal velocity of the vehicle-parachute system. For the case where the parachute is not deployed, the equations are simplified by the elimination of the drag term.

Consider a vehicle descending vertically with the following conditions:

1. Constant rocket thrust

   or $T = (T/Wo)\ Wo$

2. Drag of vehicle and chute is proportional to dynamic pressure

   $$D = K\ (v^2)$$

   At terminal velocity

   $$D = Wo = K v_T^2$$

   or $K = \dfrac{W_o}{v_T^2}$

3. Vehicle weight

$$W = Wo - \frac{dw}{dt} \, dt$$

$$= Wo - \frac{T}{Isp} \, dt$$

Assume that change of vehicle weight is small compared to initial weight.
i.e. $\frac{dw}{dt} \, dt = 0$

or $W = Wo$

The sum of the vertical forces is:

$$F = T + D - W = (T/Wo) \, Wo + \frac{Wo}{V_T^2} V^2 - Wo$$

Since $\left[F = M\frac{dv}{dt} \right]$: $(T/Wo) \; Wo + \frac{Wo}{V_T^2} \; V^2 - W_o = \frac{Wo}{g} \, \frac{dv}{dt}$

or $$T/Wo + \frac{V^2}{V_T^2} - 1 = \frac{1}{g} \, \frac{dv}{dt}$$

or $$dt = \frac{1}{g} \, \frac{dv}{T/Wo - 1 + \frac{V^2}{V_T^2}}$$

Hence $$\Delta t = \int dt = \frac{1}{g} \int \frac{dv}{T/Wo - 1 + \frac{V^2}{V_T^2}}$$

Solving for the time:

if $\underline{T/W > 1}$ $$\Delta t = \frac{1}{g} \, \frac{1}{\frac{T/Wo - 1}{V_T^2}} \, \tan^{-1} \left(V \sqrt{\frac{1}{V_T^2 (T/Wo - 1)}} \right) \Bigg]_{V1}^{V2}$$

Or if $\underline{T/W = 1}$ $$\Delta t = -\frac{1}{g} \, V_T^2 \, \left(\frac{1}{V}\right) \Bigg]_{V_1}^{V_2}$$

Using the above equations, the time and distance required to decelerate the vehicle to 10 ft./sec. are plotted as a function of rocket thrust/weight ratio (T_R/Wo) in Figure D-1. The velocity at the beginning of rocket operation was assumed equal to parachute terminal velocity.

Since the parachutes under consideration require about 150 feet of altitude for safe deployment, the deceleration distance shown in Figure D-1 should be added to this altitude to establish the minimum altitude for safe recovery using the parachute.

If an emergency occurs below this altitude the rockets should be used to maintain the same descent rate as the chute (40 ft./sec.) and the JATO arresting system operated for final deceleration.

Allowing a minimum of a 50 percent peroxide fuel margin, the above approach requires a total vehicle recovery system of 350 pounds. This includes 19 pounds for the JATO bottle and control, 265 pounds of peroxide and 66 pounds for a Stencel parachute providing a 40 ft./sec. sink rate for a 3500 pound vehicle.

For the case where the peroxide rocket is used for the final deceleration after deploying the parachute, the rocket time is 3.8 seconds and the minimum parachute deployment attitude is 220 feet.

For the case where the peroxide rocket is used instead of the parachute the initial altitude is 240 feet and the rocket time is 6 seconds.

It should be noted that this recovery system also permits an emergency technique in which a 1.1g climb of 8 seconds using all the peroxide fuel is followed by crew ejection, and then parachute and JATO vehicle recovery. Starting from a hover (the LLRV could even be on the ground) an attitude of 150 feet would be reached at the end of the climb.

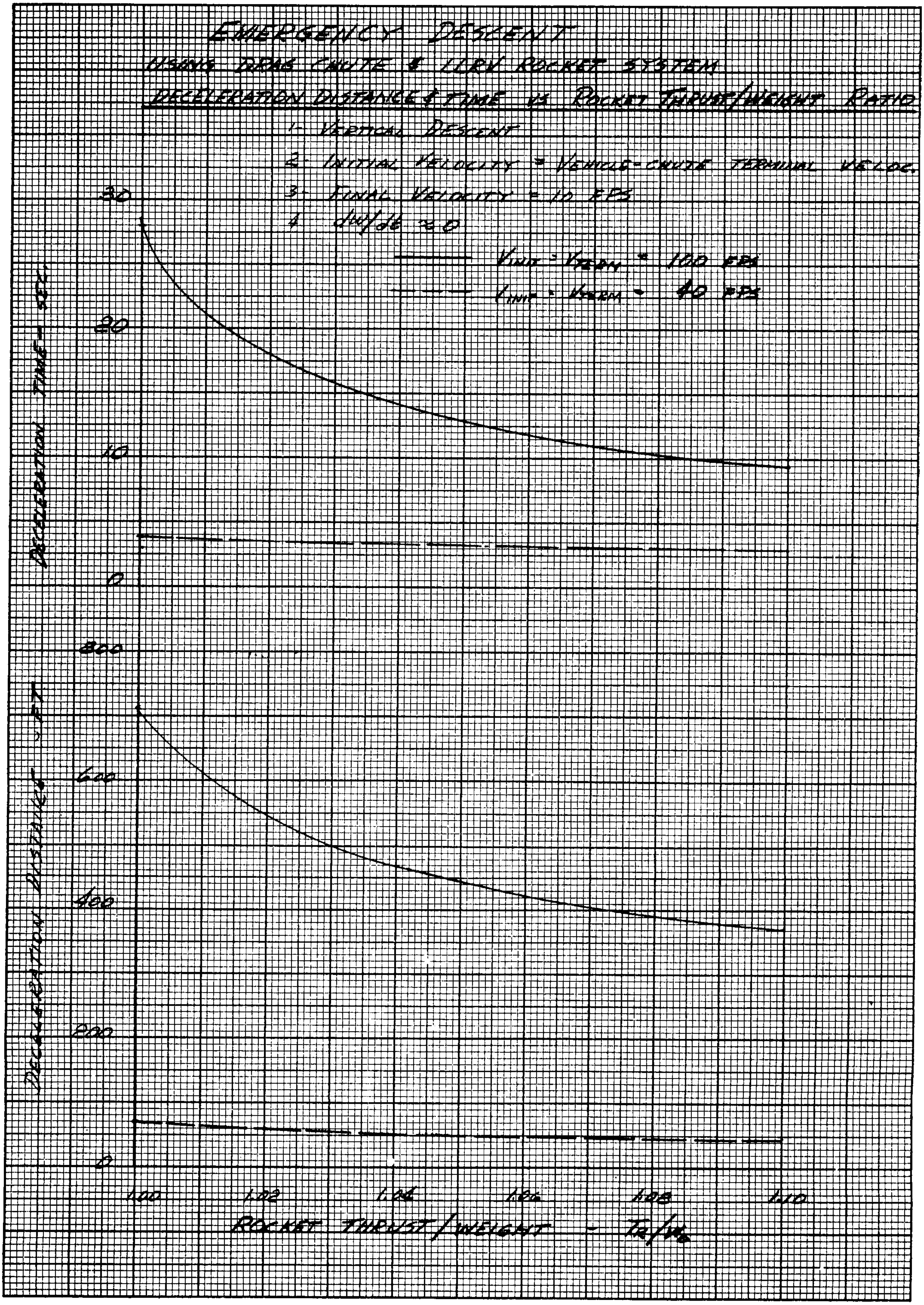
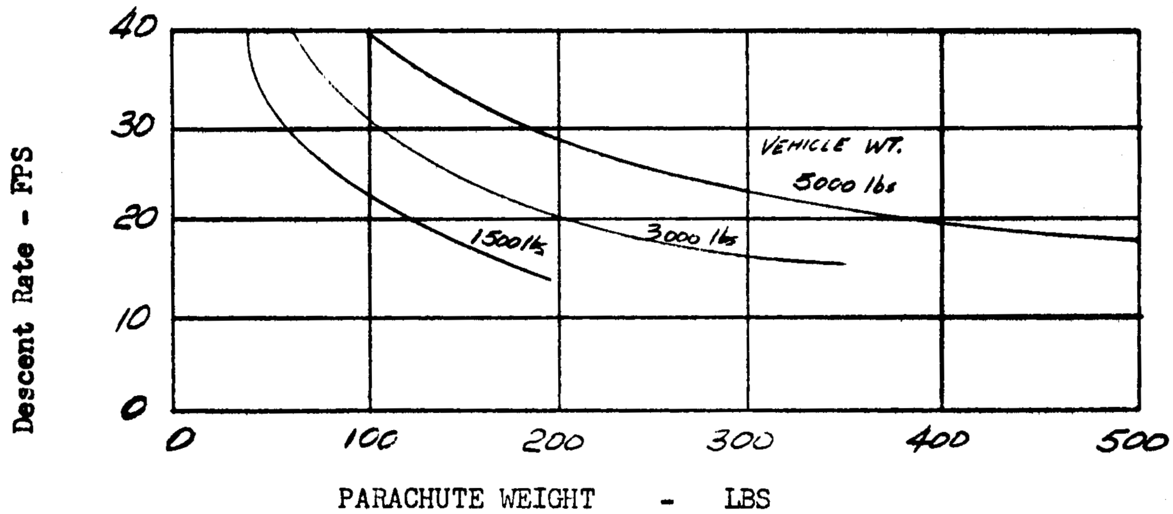

EMERGENCY DESCENT
USING DRAG CHUTE & LLRV ROCKET SYSTEM
DECELERATION DISTANCE & TIME vs ROCKET THRUST/WEIGHT RATIO
1- VERTICAL DESCENT
2- INITIAL VELOCITY = VEHICLE-CHUTE TERMINAL VELOC.
3- FINAL VELOCITY = 10 FPS
4- dw/dt ≈ 0
V_INIT = V_TERM = 100 FPS
V_INIT = V_TERM = 40 FPS
DECELERATION TIME - SEC.
30
20
10
0
DECELERATION DISTANCE - FT
800
600
400
200
0
1.00
1.02
1.04
1.06
1.08
1.10
ROCKET THRUST/WEIGHT - TR/WG

Fig. D-1

STENCEL PARACHUTE WEIGHT



For computing W_R (recovery system Weight) for vehicle weights between 1500 lbs and 5000 lbs the following equation applies:

$$W_R = 0.03\, W_L \left(\frac{30}{Ve}\right)^2 + 15$$

Descent Rate	Rec. System Weight for 3000 lb. load
40 fps	66 lbs
30	105
25	145
20	218
17	295
16	332

The above data are based on 60 kts descent for 100-150 foot altitude.

Fig. D-2

APPENDIX E

LLRV CURRENT WEIGHT STATEMENT (5 APRIL 1963)

Item	Current Weight (Pounds)
1. STRUCTURE	(510)
Leg Trusses	180
Engine Mounting Gimbals, Bearings	100
Main Platform	80
Cockpit Floor	33
Turnover Structure	34
Windshield	31
Seat Supports	17
Misc. Supports & Brackets	35
2. ALIGHTING GEAR	(154)
Pads	24
Shock Struts	84
Lord Mounts, Linkage, Misc.	46
3. CONTROLS - MANUAL	(34)
Flight	17
Engine	8
Rockets	9
4. CONTROLS - AUTOMATIC	(84)
Vehicle & Engine Stabilization, Engine Thrust	84
5. POWER PLANT	(709)
Engine (Residuals Included 3lb)	639
Air Induction	14
Fuel System (Dry)	56
Starting System	--
6. ROCKET SYSTEM (Dry)	(315)
7. INSTRUMENTS	(45)
Flight & Navigation	18
Engine	8
Rockets	7
Instrument Board & Installation	12

	Item	Current Weight (Pounds)
8.	HYDRAULICS & PNEUMATICS	(52)
	Engine Gimballing	24
	Gimbal Locking	20
	Power Supply (Hydraulic)	8
9.	ELECTRICAL	(90)
10.	COMMUNICATIONS	(10)
	Communication Set	10
11.	FURNISHINGS	(109)
	Ejection Seat (Incl. Chute & Harness)	99
	Oxygen Supply	10
12.	AUXILIARY GEAR	(57)
	Ground Handling	7
	Recovery Drogue System	50
	CURRENT WEIGHT EMPTY	2169
13.	USEFUL LOAD	(1412)
	Pilot	200
	Payload	200
	JP4 Fuel	400
	H_2O_2 Fuel	600
	Helium Gas	4
	Usable Engine Oil	8
	CURRENT GROSS TAKE-OFF WEIGHT	3581